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RDT&E PROJECT NO. ____________
USATECOM PROJECT NO. 4-4-0108-03
USAAVNTA PROJECT NO. 64-28

ENGINEERING EVALUATION OF UH-1B HELICOPTER EQUIPPED WITH MODEL 540 ROTOR SYSTEM

PHASE B

FINAL REPORT

BY

LAUREL G. SCHROERS
PROJECT ENGINEER

MICHAEL N. ANTONIOU
CAPTAIN, US ARMY TC
PROJECT PILOT

JUNE 1966

U. S. ARMY AVIATION TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA

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Photo No. 1 - UH-1B Helicopter equipped with model 540 rotor system

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ABSTRACT

An engineering evaluation (Phase B contractor's compliance tests) of the first production UH-1B helicopter equipped with the Model 540 Rotor System, S/N 63-8684, was conducted at the contractor's facilities from 26 February 1965 to 23 March 1965. The U. S. Army Aviation Test Activity (USAAVNTA) participated in this program to familiarize its personnel with the manufacturer's envelope. An additional objective was to report on test results as they compared with the manufacturer's engineering change proposal for the incorporation of the Model 540 Rotor System in the UH-1B helicopter.

The USAAVNTA was responsible for preparing test plan and coordinating with the U. S. Army Aviation Test Board, participating in contractor's compliance tests, and submitting a final report to Hq, U. S. Army Test and Evaluation Command.

The level flight performance and climb performance of the UH-1B/540 rotor helicopter were generally improved over those of a standard UH-1B. Maximum airspeed in level flight was in most cases limited by power available. Steady-state vibrations were generally considerably improved over those of a standard UH-1B. A transient self-excited rotor and pylon system oscillation, "pylon rock," was encountered in a small portion of the flight envelope and considered highly objectionable. The low high-speed static longitudinal stability was considered unacceptable. Autorotational characteristics were sufficiently different from those of a standard UH-1B as to require a pilot check-out.

It is recommended that the self-excited "pylon rock" deficiency be eliminated and the low high-speed static longitudinal stability deficiency be improved before Phase D or service testing.

Correction of several shortcomings listed in this report would result in improved helicopter capabilities.

FOREWORD

1. AUTHORITY

a. Letter, AMSTE-BG, Hq, U. S. Army Test and Evaluation Command (USATECOM), 20 August 1964, subject: "Test Directive for USATECOM Project No. 4-4-0108-03/04, Model 540 Rotor System Tests."

b. Letter, AMSTE-BG, Hq, USATECOM, 3 November 1964, subject: "Amendment to Test Directive for USATECOM Project Task Number 4-4-0108-03/04."

c. Letter, AMSTE-BG, Hq, USATECOM, 11 March 1965, subject: "Amendment to Test Directive for USATECOM Project Task Number 4-4-0108-03/04."

2. REFERENCES

A list of references is contained in Section 3, Appendix VI.

SECTION I - GENERAL

1.1 OBJECTIVES

The objective of this test was to participate in the contractor's Phase B compliance flight tests to familiarize U. S. Army Aviation Test Activity (USAAVNTA) personnel with the manufacturer's flight envelope. An additional objective was to report on test results as they compare with the manufacturer's engineering change proposal (Reference a, Section 3, Appendix VI).

1.2 RESPONSIBILITIES

The responsibilities were to prepare test plan and coordinate with the U. S. Army Aviation Test Board, participate in contractor's Phase B compliance flight tests, and report on test results to Hq, U. S. Army Test and Evaluation Command (USATECOM).

1.3 DESCRIPTION OF MATERIEL

The UH-1B helicopter is being procured by the Army as a general utility helicopter to transport personnel and equipment. Various armament kits are available for installation on UH-1B helicopters equipped with the required hard points. Engineering Change Proposal ECP-UH-1B-160 (Reference a, Section 3, Appendix VI), which provided for the incorporation of the Model 540 Rotor System in production UH-1B helicopters beginning in August 1965, was procured and incorporated in the test helicopter. A second Engineering Change Proposal, ECP-UH-1B-161 (Reference b), which provided for an increase in usable UH-1B fuel capacity from 1008 pounds to 1573 pounds, was procured in conjunction with ECP-UH-1B-160 but was not incorporated in the test helicopter. This engineering change proposal will be incorporated in future UH-1B helicopters.

The Model 540 rotor is a two-bladed teetering rotor system with a 44-foot diameter, the same as the diameter of the rotor used on earlier UH-1B helicopters. In all other respects, the Model 540 rotor is considerably different, as shown in Table 1.

TABLE 1. DIFFERENCES BETWEEN STANDARD UH-1B ROTOR AND MODEL 540 ROTOR

	UH-1B Standard Rotor	UH-1B Model 540 Rotor
Rotor Diameter-ft	44	44
Chord-in	21	27
Twist-deg	-10	-10
Airfoil	NASA 0012	Special 0009 1/3
Disc Area-sq ft	1520	1520
Blade Area-sq ft per blade	38.5	49.5
Solidity Ratio	.0506	.0651
Rotational Inertia-slugs/ft^2	1660	2800

A detailed description of the UH-1B/540 rotor helicopter's configuration as delivered to USAAVNTA is presented in Section 3, Appendix IV.

1.4 BACKGROUND

In October 1963, the contractor proposed a 20-hour flight evaluation of the 540 "Door Hinge" Rotor System at no cost to the Government. In November 1963, the Office of the Chief of Research and Development, Department of the Army, requested USAMC to accept the proposal. USAMC assigned the flight evaluation to USATECOM in August 1964. The 20-hour flight evaluation was conducted by USAAVNTA during the period 8 January 1964 to 22 January 1964. A model 204B helicopter, civilian version of the Army's UH-1D helicopter, was used for this test. The 204B differed from the UH-1B principally in the incorporation of an extended tail boom to allow operation with a 48-foot rotor. Based on the results of this evaluation (Reference c, "Military Potential Test of the Model 540 'Door Hinge' Rotor System) the contractor's Engineering Change Proposal ECP-UH-1B-160 was procured (Reference a). As a result of this procurement, the Model 540 Rotor System became standard on the production UH-1B helicopters beginning in August 1965.

On 12 August 1964, the Iroquois Project Manager, USAMC, requested USATECOM to conduct Phase B and Phase D tests of the UH-1B helicopter equipped with the Model 540 Rotor System, as required by AR 70-10. In Test Directive, 20 August 1965, amended 3 November 1964 and 11 March 1965, USATECOM authorized USAAVNTA to conduct Phase B and Phase D tests. USAAVNTA submitted a Plan of Test which was approved and forwarded by USATECOM to USAMC on 18 February 1965.

Phase B tests were conducted at the contractor's facilities during the period 26 February 1965 to 23 March 1965. A total flight time of 47.4 hours was accumulated during this period. Approximately 35 hours were required to conduct Phase B tests and approximately 12.4 additional hours were required to define problem areas that had been uncovered during Phase B. USATECOM, on 23 April 1965, requested USAAVNTA to evaluate the contractor's corrections to problem areas uncovered in Phase B. This required that USAAVNTA participate in the contractor's Phase C design refinement tests. The results of this Phase C evaluation are presented in Appendix V. USATECOM concurred in the conclusions and recommendations contained in the USAAVNTA Phase C evaluation.

Interim reports based on Phase B and Phase C test results (References h and j) were submitted to USATECOM by USAAVNTA on 31 March 1965 and 5 May 1965 respectively.

1.5 FINDINGS

See Section 2 for a full discussion of test findings.

1.6 Conclusions

1.6.1 Performance

a. Compared with the standard UH-1B, the UH-1B/540 rotor helicopter as tested had a decrease in climb performance below 9800 feet. Above this altitude an increase in climb performance with a corresponding increase in service ceiling was realized. (Paragraph 2.3.4.1)

b. Phase B level flight performance data agrees with the prototype 540 rotor data presented in USAAVNTA Report ATA-TR-64-2 (Reference c). (Paragraph 2.4.4.1)

c. Although the UH-1B/540 rotor helicopter had a higher maximum speed in level flight than the standard UH-1B, the range performance was not markedly different. (Paragraph 2.4.4.1 b)

d. The maximum level flight speeds of the UH-1B/540 rotor helicopter as tested were power limited at all gross weights and altitudes. (Paragraph 2.4.4.1 c)

e. Autorotational descent performance of the UH-1B/540 rotor helicopter tested agreed favorably with the prototype 540 autorotation data presented in USAAVNTA Report ATA-TR-64-2 (Reference c). (Paragraph 2.5.4.1)

f. The standard ship airspeed system had a negative position error at all speeds in level flight above 40 KIAS with a maximum error of 4 knots at 140 KIAS. (Paragraph 2.7.4)

g. A variation in position error of 3 to 6 knots during climbing flight within the airspeed range for maximum rate of climb was undesirable. (Paragraph 2.7.4)

1.6.2 Vibrations

a. Steady-state vibration levels were generally within the limits of Paragraph 3.7.1, MIL-H-8501A (Reference 1) at all conditions that could be achieved during power-limited flight and agreed with prototype 540 rotor vibration data published in USAAVNTA Report ATA-TR-64-2 (Reference c). Moderate 1-per-rev vibration levels were experienced in low-power descents, higher power climbs and during transient maneuvers. Exceptionally low 2-per-rev vibration levels were encountered. (Paragraph 2.4.4.2)

b. The rotor and pylon system low-frequency oscillation characteristics were unsatisfactory. A self-excited oscillation, identified as undamped pylon motion, was experienced. This produced a circular lateral-longitudinal motion with a superimposed vertical amplitude as high as plus or minus .4 of a g with a frequency of approximately 3.0 cycles per second (cps). The normal characteristics of this oscillation were that it was self-excited, built up to maximum level very rapidly and appeared to be neutrally damped. Pilot-induced pylon and rotor oscillations were most noticeable during maneuvering flight when control inputs were rapid and frequent. This resulted in a continued state of oscillatory motion that detracted from the pilot's tactical effectiveness. (Paragraph 2.4.4.2)

1.6.3 Stability and Control

a. The longitudinal cyclic control position with an aft center of gravity (C.G.) was uncomfortable at the light-weight, low-altitude, power-limit airspeeds. The extreme forward position of the longitudinal cyclic control would make extended flight under these conditions fatiguing. (Paragraph 2.6.4.1)

b. The apparent speed stability of the UH-1B/540 rotor helicopter was satisfactory with a forward C.G. As the C.G. was moved aft, however, the longitudinal cyclic control position gradient became more shallow but still met the requirements of Paragraph 3.2.10, MIL-H-8501A (Reference 1). (Paragraph 2.6.4.2)

c. Stick-free static longitudinal stability was qualitatively evaluated as satisfactory up to 60 knots calibrated airspeed (KCAS). Above this speed the force gradient became nonlinear and less force was required to increase airspeed than to decrease airspeed about a trim point. (Paragraph 2.6.4.3)

d. Stick-fixed static longitudinal stability was satisfactory at speeds up to 100 KCAS. Above this speed the stability with respect to stick position gradient became quite shallow up to 140 KCAS. This condition, in itself, was considered satisfactory; however, when coupled with the loss of force gradient due to the loss of position gradient, which was a result of the force trim design, an unsatisfactory condition existed. (Paragraph 2.6.4.4)

e. Trim authority was satisfactory at high speed with an aft C.G. Trim authority in rearward flight with a forward C.G. was unsatisfactory. The contractor had not provided a cyclic trim system that allowed cyclic control forces to be trimmed out in accordance with the requirements of Paragraph 3.2.3 of MIL-H-8501A (Reference 1). (Paragraph 2.6.4.5)

f. Handling qualities during entry into autorotation following a simulated engine failure at high speed were unsatisfactory. Throttle chops at high speed caused the helicopter to roll left and pitch down excessively and required a 3-inch right, 1-inch aft cyclic trim change. This is an undesirable characteristic that was not present in the prototype 540 rotor hardware. (Paragraph 2.6.4.6)

g. Control of rotor speed during autorotation was more difficult than in previous UH-1 series helicopters. Increased inertia of the 540 rotor system produced a response to collective inputs different from that of a standard UH-1B. (Paragraph 2.5.4.2)

h. Stabilized autorotational descents at 40 knots indicated airspeed (KIAS) were difficult because a low-frequency, large-amplitude yawing oscillation was experienced. Satisfactory autorotational handling qualities were experienced in stabilized descents above 60 KIAS. (Paragraph 2.5.4.2)

i. The autorotational landing technique required for the UH-1B/540 rotor helicopter was sufficiently different from that of the standard UH-1B as to require a pilot check-out. Maximum skid height for hovering autorotations in ground effect (IGE) was determined to be 5 feet for all gross weights. (Paragraph 2.5.4.2)

1.7 Recommendations

a. Correction of the following deficiencies must be accomplished prior to Phase D or service testing:

(1) The self-excited "pylon rock" characteristic must be eliminated. (Paragraph 2.4.4.2)

(2) The low high-speed static longitudinal stability must be improved. (Paragraph 2.6.4.3)

b. Correction of the following shortcoming will result in a helicopter of improved capabilities:

(1) Inadequate trim authority as required by Paragraph 3.2.3 of MIL-H-8501A (Reference 1). (Paragraph 2.6.4.5)

(2) Excessive trim change when transitioning from high-speed powered flight to autorotation. (Paragraph 2.6.4.6)

(3) Uncomfortable control position at high speed with an aft C.G. (Paragraph 2.6.4.1)

(4) Poor control force harmony. (Paragraph 2.6.4.7)

(5) Excessive 1-per-rev vibration level experienced in high-power climb, low-power descents and during transient maneuvers. (Paragraph 2.4.4.2)

SECTION 2. DETAILS OF TEST

2.0 INTRODUCTION

Test results were compared with those claimed in the contractor's Engineering Change Proposal ECP-UH-1B-160 (Reference a). Test data was also compared with data in USAAVNTA Report ATA-TR-64-2, "Military Potential Test of the Model 540 'Door Hinge' Rotor System" (Reference c). Stability and control data was evaluated on the basis of requirements of Military Specification MIL-H-8501A (Reference 1).

2.1 HOVER

2.1.1 Objective

The objective of these tests was to evaluate handling qualities and power management characteristics while hovering in ground effect (IGE) and out of ground effect (OGE) and while transitioning from IGE to OGE. Handling qualities and power management characteristics were also investigated in sideward and rearward flights conducted IGE.

2.1.2 Method

Hovering performance tests as outlined in the plan of test (Reference e) were not accomplished because calm wind weather conditions were not available during the Phase B test period. Hovering tests were conducted at various gross weights and centers of gravity (C.G.'s) in winds to evaluate handling qualities with the force trim system both on and off.

2.1.3 Results

Quantitative hovering performance results were not obtained. Qualitative results are discussed in the following paragraphs.

2.1.4 Analysis

2.1.4.1 Handling Qualities

During engine start the hydraulic control boost became effective at approximately 30-percent gas producer speed (N_1); this was satisfactory.

Lift-off to a hover at the most adverse conditions (forward and aft limit C.G.'s) was easily and satisfactorily accomplished with only small cyclic trim changes required. Lift-off at 9680 pounds and a forward C.G. (Station 126.5 inches) was accomplished with approximately 1.5-inch aft cyclic displacement required.

Handling qualities while hovering "over a spot" were satisfactory and control was maintained with only small cyclic control inputs.

It appeared that sufficient control power was available to obtain easily 35-knot sideward flight and 30-knot rearward flight. The normal pedal reversal characteristic of UH-1B's in left sideward flight that occurs at approximately 8-10 knots appeared to be greater than for the standard UH-1B. As stated in Paragraph 2.1.2, the weather was not ideal; therefore, control positions in sideward and rearward flight must be fully evaluated during Phase D.

Cyclic trim authority was inadequate in rearward flight. An aft cyclic control force of approximately 5 pounds was required to hover with a 25-knot tailwind. The contractor had not provided a cyclic trim system that allowed cyclic control forces to be trimmed out in accordance with the requirements of Paragraph 3.2.3 of MIL-H-8501A (Reference 1).

Collective control loads in hover and cruise flight were measured and evaluated as satisfactory. The contractor incorporated a minimum collective friction of 8-10 pounds; this was later evaluated as satisfactory for cruise flight but a little too heavy for extended hovering (see Phase C Test Results, Paragraph 1.3 f, Section 3, Appendix V). This will be evaluated further in Phase D.

A transient nose-up trim change was experienced when going from an IGE hover condition to an OGE hover condition. The steady-state trim change was approximately 0.5-inch forward cyclic control displacement.

2.1.4.2 Power Management

Engine start and acceleration to operating speeds were accomplished without difficulty and throttle minimum friction level was satisfactory.

Lift-off to a hover usually produced a 4-rpm droop in rotor speed. A lift-off at 9680 pounds was accomplished at a density altitude of 1150 feet; however, the subsequent control motions required to hover produced a power requirement which was greater than power available. In this condition, rotor speed drooped below the 300-rpm power-off minimum limit and continued to droop until the helicopter settled to the ground. This droop occurred with full beep control (power turbine speed selector) applied. The contractor's ECP-UH-1B-160 (Reference a) indicated a 45-percent increase in mission productivity because the maximum allowable gross weight had been increased to 9500 pounds. This 45-percent increase in mission productivity was not realized because the helicopter became power-limited at 9500 pounds and atmospheric conditions close to sea-level standard day.

There was a lag of approximately 1.0 to 1.5 seconds between application of the beep control and an indicated change in rpm. This was satisfactory but should be improved.

2.2 TAKEOFF

2.2.1 Objective

The objective of these tests was to evaluate qualitatively handling qualities, power management and vibration characteristics during takeoff.

2.2.2 Method

Special takeoff tests were not conducted. Takeoff handling qualities, power management and vibration were qualitatively evaluated at the beginning of each test flight. The following three takeoff techniques were used:

a. Two-foot level acceleration - Takeoff was initiated from a 2-foot hover and the helicopter was accelerated to a desired climbout airspeed while maintaining a 2-foot skid height and constant rotor speed.

b. Climb and acceleration - Takeoff was initiated from a light-on-the-skids condition. As takeoff power was applied and lift-off occurred, a pitch attitude was selected and held constant to obtain a desired airspeed at 50 feet.

c. Sliding - Takeoff was initiated from a light-on-the-skids condition by application of forward cyclic control. Ground contact was maintained until sufficient translational lift was available to achieve lift-off and level acceleration. This takeoff was used when insufficient power was available to hover.

2.2.3 Results

Quantitative results were not obtained; however, qualitative results of handling qualities, power management and vibration are discussed in the following paragraphs.

2.2.4 Analysis

2.2.4.1 Handling Qualities

The test helicopter exhibited satisfactory handling qualities using all three takeoff techniques. Sufficient control power was available to produce the desired pitching moments necessary to execute accurately the level acceleration and the climb and acceleration techniques. Normally, the most critical takeoff technique as far as handling characteristics are concerned is the sliding takeoff. Using this technique, a takeoff was accomplished from a soft, grassy area. Acceleration to lift-off airspeed was constant and a nose-low attitude was maintained without excessive control manipulations until lift-off airspeed was achieved.

2.2.4.2 Power Management

Takeoffs were accomplished using the three techniques described in Paragraph 2.2.2. Power management was not a problem. The beep switch lag described in Paragraph 2.1.4.2 was again annoying but not unsatisfactory.

2.2.4.3 Vibration Characteristics

The vibration characteristics experienced during takeoff were random in nature and of mixed frequencies. The most noticeable vibrations were experienced while passing through translational lift, at rotation and during initial climbout. These vibration characteristics were annoying but not unsatisfactory.

2.3 CLIMB

2.3.1 Objective

These tests were conducted to determine the rate of climb and service ceiling. An additional objective was to evaluate qualitatively handling qualities, vibration, and power management in climbing flight.

2.3.2 Method

Two continuous climbs were flown using maximum power available from 2000 feet to service ceiling. These flights were flown in non-turbulent air with a mid C.G. and a gross weight of 7660 pounds. This weight was used so that a comparison of previously published data could be made. The climb speed schedule used was calculated from level flight data presented in USAAVNTA Report ATA-TR-64-2 (Reference c).

2.3.3 Results

Climb results are presented graphically in Figure 1, Section 3, Appendix I.

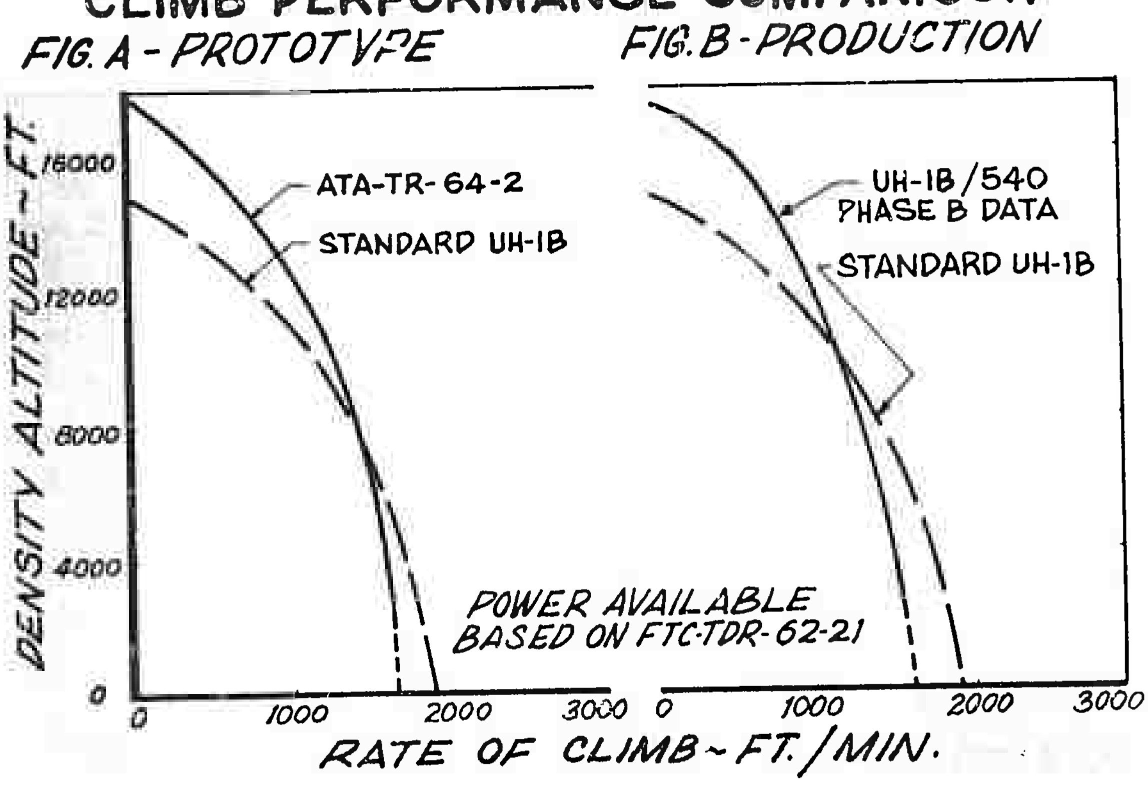
2.3.4 Analysis

2.3.4.1 Performance

The UH-1B/540 rotor helicopter, compared to the standard UH-1B, had a decrease in climb performance below 9800 feet. Above this altitude the UH-1B/540 rotor helicopter exhibited an increase in climb performance with a significant increase in service ceiling.

The relatively small decrease in climb performance (approximately 300 feet per minute) at low altitude was undesirable but was a rotor design compromise. The increase in service ceiling (approximately 3000 feet or 16.5 percent) was a significant increase and a desirable improvement.

CLIMB PERFORMANCE COMPARISON



Figures A and B present a comparison of climb performance of the prototype 540 rotor system and the production 540 rotor system. From Figure A it can be seen that the crossover in climb performance of the prototype, compared to a standard UH-1B, occurs at 6500 feet. With the production 540 rotor this crossover occurs at 9800 feet (Figure B). The climb performance curves for both cases are based on power available as presented in Report FTC-TDR-62-21 (Reference m).

The climb speed schedules used for the prototype 540 and the production 540 rotor are presented in Figure C. The climb speed schedule used for the production 540 rotor tests (Phase B) was calculated from level flight performance of the prototype 540 rotor presented in USAAVNTA Report ATA-TR-64-2 (Reference c). The variation in climb speed schedules does not explain the difference between the climb performance of the prototype and the production 540 rotors because at sea level, where the greatest difference in climb speed schedules existed, the rate of climb of the production 540 compared favorably with that of the prototype 540. At 10,500 feet, where the climb speed schedules agreed, the difference in rates of climb was greater. Also presented in Figure C is the climb speed schedule calculated from Phase B data on the production 540 rotor system. This shows that the climb speed schedule used for Phase B tests was approximately 5 knots too fast. This was also the qualitative opinion of the pilot and was most noticeable at or near the service ceiling. Sawtooth climbs will be conducted during Phase D tests to obtain an optimum climb speed schedule before climb performance is obtained for inclusion in the Operator's Manual (Reference p).

CLIMB PERFORMANCE COMPARISON

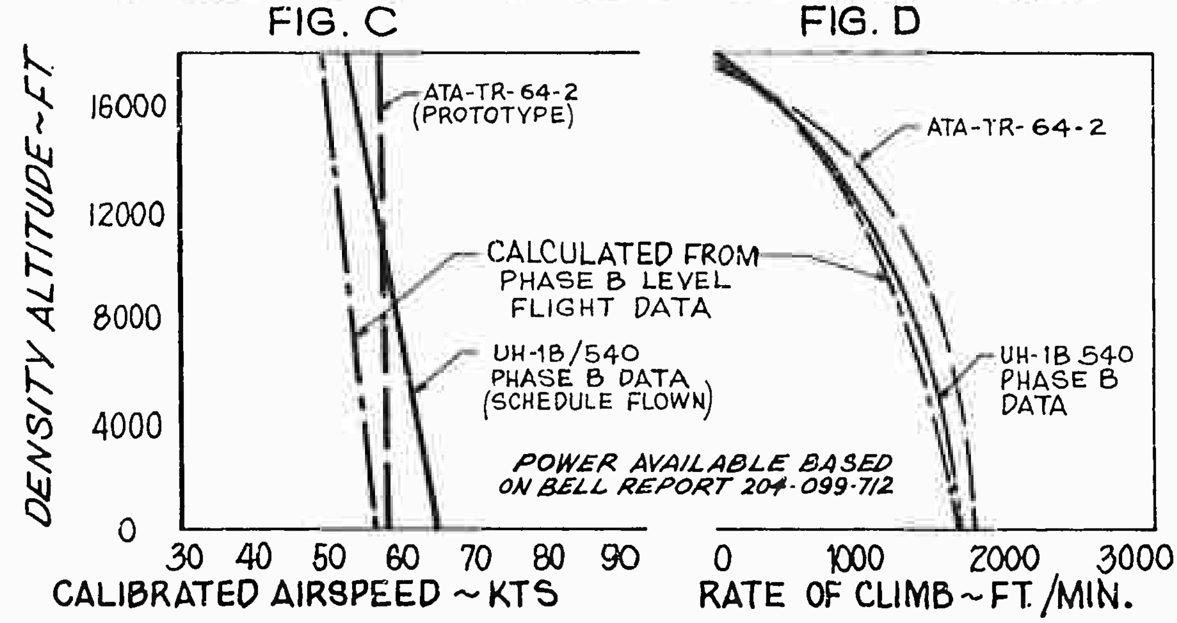


Figure D presents the climb performance of the production and prototype 540 rotor systems based on power available from Reference n. This is the power available that will be used for preparation of the UH-1B/540 rotor Operator's Manual. Also shown in Figure D is the rate of climb calculated from Phase B data; this agrees favorably with the actual Phase B climb data.

2.3.4.2 Handling Qualities

The UH-1B helicopter equipped with the 540 rotor system had satisfactory handling qualities throughout the climb from sea level to service ceiling. A small nose-down trim change occurred within the altitude range of 6000 to 8000 feet. This resulted in an increase in airspeed of 4 to 5 knots; this increase required aft cyclic displacement to maintain the desired climb speed schedule.

Flying qualities in maneuvering flight at the service ceiling were excellent. The normal decrease in damping about the pitch and roll axes that occurs with increasing altitude was not apparent to the pilot and the control sensitivity about the pitch and roll axes was satisfactory.

2.3.4.3 Power Management

Once power was established at maximum available, the climb speed schedule could be held without difficulty. Maximum power was established by increasing collective pitch (with full beep) until rotor rpm started to droop. Collective pitch was then adjusted throughout the climb to maintain the desired rotor speed. During the two continuous climbs the helicopter was power-limited; that is, at no time was a torque, exhaust gas temperature or gas producer speed limit encountered.

2.3.4.4 Vibrations

The vibration characteristics during climb were qualitatively evaluated as satisfactory but they could be improved. Vibration characteristics were random in nature and predominantly 1 per rev.

The contractor attributed the 1-per-rev vibration to the fact that the blade profile at the inboard sections was not within manufacturing tolerances.

2.4 LEVEL FLIGHT

2.4.1 Objective

These tests were conducted to determine:

a. Performance

(1) Power required versus airspeed for variations in altitude, gross weight, C.G. and rotor speed within the flight envelope.

(2) Range factor and recommended cruise speed for variations in altitude, gross weight, C.G. and rotor speed within the flight envelope.

(3) Maximum level flight airspeed with variations in gross weight and altitudes.

b. Power management with variations in gross weight and altitude.

c. Vibration characteristics

(1) Steady-state vibrations

(2) Rotor and pylon system oscillations

2.4.2 Method

Level flight performance tests, (speed-power polars) were conducted in non-turbulent air at a constant value of thrust coefficient (C_T). A constant C_T was maintained by flying at higher density altitudes as fuel was consumed.

Five speed-power polars were flown for baseline data at the maximum power on rotor speed of 324 rpm and with a mid C.G. The altitude and gross weight conditions for these five polars were selected to produce C_T values that covered the major portion of the UH-1B/540 rotor helicopter performance range.

Two speed-power polars were flown to determine the effect of C.G. on power required for level flight. These polars were flown at approximately 7500 pounds gross weight, 5000 feet density altitude, one each at a forward and an aft C.G.

Two speed-power polars were flown to determine the effect on performance of flight at the minimum power on rotor speed of

314 rpm. These polars were flown at 10,000 feet with a mid C.G., and at gross weights of 6500 pounds and 7500 pounds.

2.4.3 Results

A summary of level flight performance test results is presented graphically in Figure 2, Appendix I. Results of the five speed-power polars flown for baseline data were reduced to non-dimensional parameters. From this data, curves of power coefficient (Cp) versus thrust coefficient (C_T)were plotted for constant rotor advance ratios (μ). The results are presented graphically in Figures 3 through 6. These non-dimensional curves were then used to obtain curve fairings for the individual speed-power polars and to determine changes in performance caused by C.G. They also provided a means of comparing the performance of the UH-1B/540 rotor helicopter with that of the standard UH-1B helicopter. Results of the individual speed-power polars are presented graphically in Figures 7 through 13.

Steady-state vibration data was recorded on all speed-power points. Because of the time frame allotted to this program, only selected vibration data was reduced to determine vibration characteristics with changes in C.G. and gross weight (Figures 14 through 25). Rotor and pylon system oscillations characteristic waveforms were recorded and analyzed at the test site and are discussed in detail in Paragraph 2.4.4.2.

2.4.4 Analysis

2.4.4.1 Performance

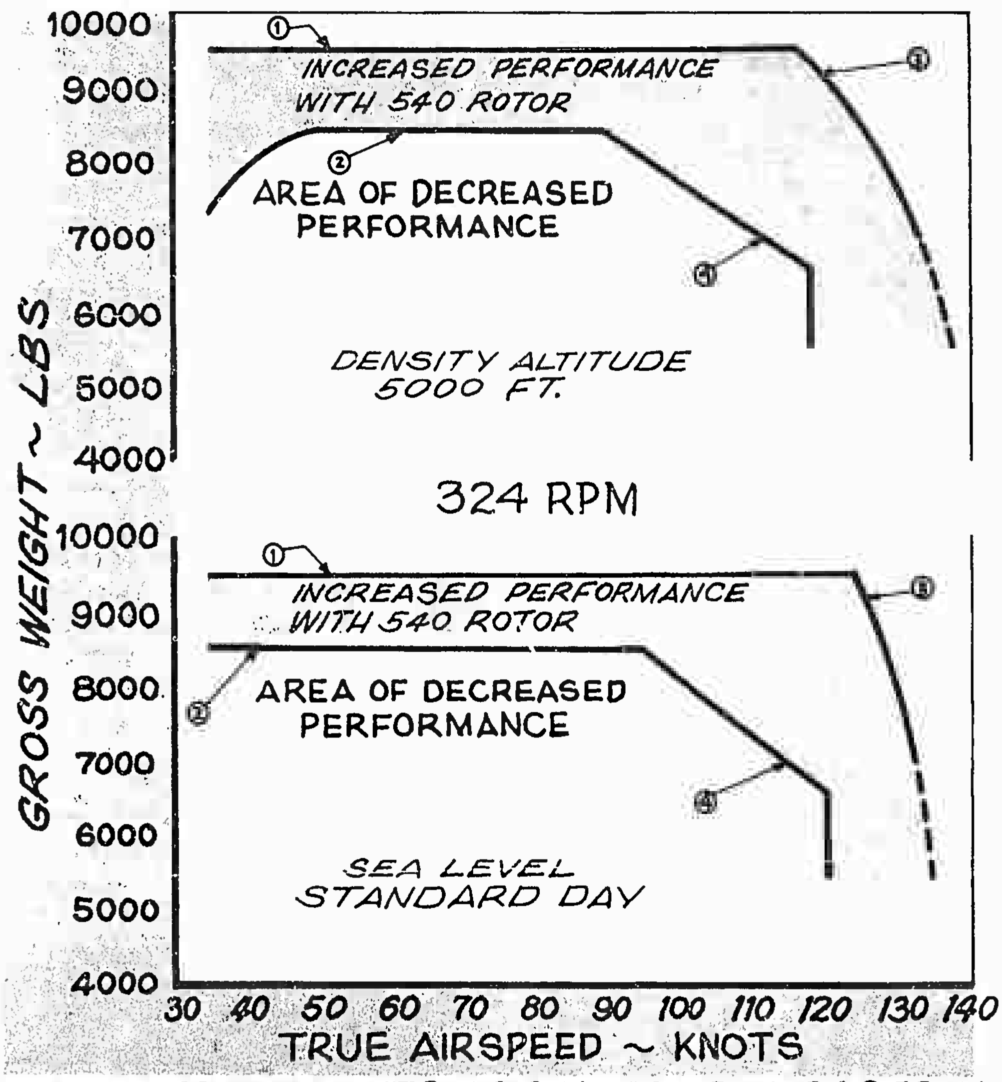
a. Power Required

Phase B level flight performance data agrees favorably with the prototype 540 rotor data presented in USAAVNTA Report ATA-TR-64-2 (Reference c). As is the case with the prototype 540 rotor, the production UH-1B/540 rotor data (Figures 3 through 6) shows a significant reduction in power required at the higher C_T values (heavy gross weight and/or high altitude) as compared with test results of a standard UH-1B helicopter. At low C_T values (light gross weight and/or low altitude) power required is greater than for a standard UH-1B helicopter.

Figure E illustrates the gross weight, altitude and airspeed required before an increase in performance was realized with the production 540 rotor system. The production

FIG. E - LEVEL FLIGHT PERFORMANCE COMPARISON

UH-1B VERSUS UH-1B/540



__NOTE__: LIMITS OF SHADED AREA BASED ON:

1. *UH-1B/540 GROSS WEIGHT LIMIT*
2. *UH-1B GROSS WEIGHT LIMIT*
3. *UH-1B/540 POWER LIMIT AIRSPEED*
4. *UH-1B PLACARD LIMIT AIRSPEED*

UH-1B/540 rotor helicopter had to be operated as indicated within the shaded area before a performance increase was realized. Several factors must be considered when looking at this area of increased performance. First, the empty gross weight of the production UH-1B/540 rotor helicopter has been increased approximately 263 pounds as compared with a standard UH-1B. This reduces the size of the shaded areas of Figure E a corresponding amount. Second, the future production UH-1B/540 rotor helicopter will have an increase in fuel capacity of approximately 500 pounds (Reference b). Fuel load versus payload, therefore, must be considered for each mission to realize the greatest increase in performance.

The standard UH-1B helicopter has a placard limit airspeed imposed by structural and vibration restrictions that is less than the power-limit airspeed of the UH-1B/540 rotor helicopter. The increased speed capability of the UH-1B/540 rotor helicopter accounts for a major portion of the increased performance capability and the gross weight increase is secondary.

Figures 12 and 13, Appendix I show the effect of C.G. on power required for a level flight condition of 7500 pounds gross weight and 5000 feet altitude. At the recommended cruise speeds, a forward limit C.G. increased, and an aft limit C.G. decreased, power required by 31 shaft horsepower (SHP). This corresponded to a change in equivalent flat plate area of approximately 2.5 square feet.

The results of level flight tests conducted at a minimum power on rotor speed of 314 rpm were inconclusive. Additional data is required before the effect of rotor speed on performance can be determined. This will be obtained during Phase D testing.

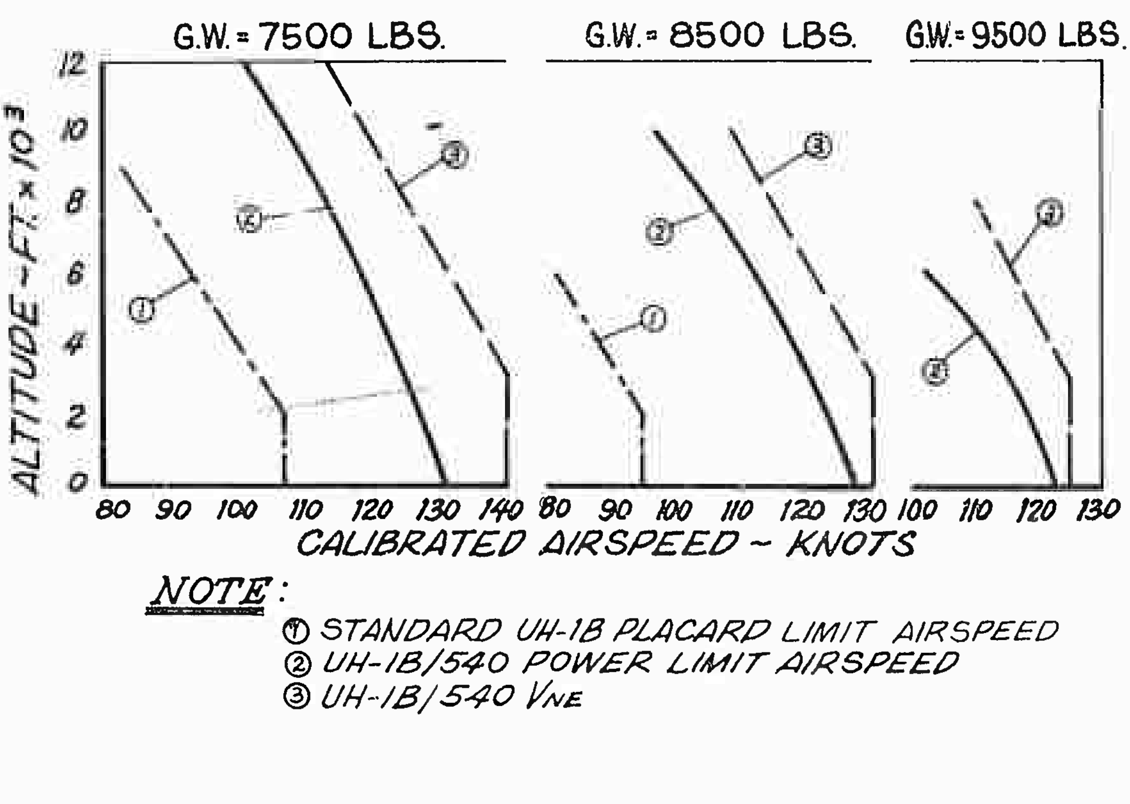
b. Range Factor and Cruise Speed

Although the UH-1B/540 rotor helicopter had a higher maximum speed than a standard UH-1B, the range performance was not markedly different (Figure 2). Analysis of Phase B test data revealed that at C_T values below .00483 the range performance was less than that of a standard UH-1B. This C_T value corresponded to a 9740-pound gross weight at sea level when operating at 324 rotor rpm. The cruise speed at this C_T value, however, increased approximately 7.5 knots with the 540 rotor system. Also, the cruise speed at optimum C_T for range of the UH-1B/540 rotor helicopter was 7 knots greater than the cruise speed of the standard UH-1B at its optimum C_T. The recommended cruise speed of the production UH-1B/540 rotor helicopter was 2 knots higher than the recommended cruise speed of the prototype UH-1B/540 rotor helicopter presented in USAAVNTA Report ATA-TR-64-2 (Reference c).

c. Maximum Airspeeds

The maximum airspeeds of the UH-1B/540 rotor helicopter were power-limited at all gross weights and altitudes tested. Figure F presents calculated power-limit airspeeds for various gross weight and altitude standard-day conditions. These power-limit airspeeds were calculated from Phase B level flight performance presented in Figures 3 through 6 and were based on power available from Reference n. As shown in Figure F, these power-limit airspeeds were all less than the velocity never exceed (V_{ne}) limits, but greater than the placard limit airspeed of the standard UH-1B helicopter. Airspeeds equal to V_{ne} plus 10 percent were easily obtained by using maximum power and a slight dive. The absence of a noticeable vibration increase with airspeed made it very easy to exceed inadvertently the V_{ne} limits.

FIG. F - AIRSPEED LIMITS
UH-1B/540 VS. STANDARD UH-1B



2.4.4.2 Vibrations

a. Steady-State Vibrations

Steady-state vibration levels were generally within the limits of Paragraph 3.7.1 of MIL-H-8501A (Reference 1) at all conditions that could be achieved during power-limited level flight. Phase B vibration data compares very well with vibration data obtained from the prototype 540 rotor system and is a considerable improvement compared with data of a standard UH-1B helicopter. Steady-state vibration characteristics were analyzed at frequencies of 1 per rev, 2 per rev and 4 per rev for a forward, mid and an aft C.G. with a medium gross weight (Figures 14 through 22). These frequencies were also analyzed for a heavy gross weight and mid C.G. (Figures 23 through 25).

b. Rotor and Pylon System Oscillations

The rotor and pylon system oscillations ("pylon rock") are divided into two catagories: self-exciting and pilot-induced.

The self-excited pylon system oscillation characteristic was experienced within a very small flight envelope but was highly unsatisfactory and detrimental to the UH-1B helicopter mission. This was a source of great concern during Phase B testing.

This self-excited oscillation, identified as "pylon rock," was experienced at 32 pounds per square inch (psi) engine torque pressure within the altitude-airspeed envelope of 9000 to 11,000 feet and 95 to 100 KIAS. The fuselage motion at the pilot and copilot stations was a circular lateral-longitudinal motion with a superimposed vertical component as high as plus or minus .4 of a g with a frequency of approximately 3.0 cycles per second (cps). This oscillation was experienced in perfectly smooth air and was self-excited.

The frequency of this oscillation (3.0 cps), however, was such that the pilot tended to amplify the oscillation once it was started. The normal oscillation characteristics were that it was self-excited, built up to maximum level very rapidly and appeared to be neutrally damped. The pilot was not able to induce this phenomenon as pulse-type control inputs produced a different vibration characteristic. This characteristic will be discussed later in this report. The tendency to enter into this self-excited vibration appeared to be aggravated by flying in a sideslip within the small flight envelope described. It was learned from subsequent testing that it was possible to get out of this self-excited oscillation by reducing engine torque pressure 2 pounds per square inch (psi) or by increasing or decreasing airspeed. Changing rotor rpm, however, had little effect on this oscillation condition.

The cause and the mandatory fix incorporated by the contractor to solve this "pylon rock" problem is discussed in "Phase C Test Results," Appendix V.

The rotor and pylon suspension system oscillations following a step-type control input were of high amplitude and damped in 3 to 4 cycles, followed by a lightly damped low-amplitude oscillation that persisted for 10 to 12 cycles. This pilot-induced oscillation was most noticeable during maneuvering flight when control inputs were rapid and frequent. This resulted in a continual state of oscillation that would detract from a pilot's tactical effectiveness. The g loads experienced during maneuvering flight combined with this pilot-induced oscillation caused the collective pitch control to drop; this increased pilot workload. The overall pilot qualitative evaluation of this characteristic was that the helicopter pylon support felt "loose" and should be improved. Additional data on this characteristic is presented in "Phase C Test Results," Appendix V.

2.5 AUTOROTATION

2.5.1 Objective

The objective of these tests was to obtain qualitative and quantitative data on autorotational descent performance, helicopter and rotor handling qualities in autorotation and autorotational landing characteristics.

2.5.2 Method

One continuous autorotational descent was conducted to obtain quantitative data at a 7400-pound gross weight, mid C.G. and 323 rotor rpm for an altitude range of 14,000 feet to 3000 feet (Figure 26).

Numerous autorotational descents were conducted in conjunction with other test flights to evaluate qualitatively helicopter and rotor handling qualities. These tests were conducted at airspeeds of 40 to 120 KIAS, gross weights from

6200 pounds to 9500 pounds, with various C.G.'s and rotor speeds from 300 to 339 rpm.

Autorotational landings were accomplished at gross weights of 6600 pounds with an aft C.G. and 9500 pounds with a forward C.G.

2.5.3 Results

Quantitative autorotational descent performance test results are presented graphically in Figure 26. This data, and qualitative data, are discussed and analyzed in the following paragraphs.

2.5.4 Analysis

2.5.4.1 Descent Performance

The quantitative autorotational descent performance data of the production UH-1B/540 helicopter agrees favorably with the autorotational descent performance data obtained from the prototype 540 rotor system and presented in USAAVNTA Report ATA-TR-64-2 (Reference c). The descent performance was satisfactory for all conditions tested, and no excessive rates of descent were encountered.

2.5.4.2 Handling Qualities

The helicopter and rotor handling qualities of the UH-1B/540 rotor helicopter were considerably different from those of a standard UH-1B. A reason for this change in handling qualities was the increased inertia of the 540 rotor system which produced a different response characteristic following collective control input. There was an increased lag in response following a collective control input. It was not difficult to keep the rotor speed within limits during a stabilized autorotation; however, it was difficult to hold a precise rotor rpm and airspeed as required to obtain conditions that produced minimum rate of descent or minimum angle of descent.

Entry into a practice autorotation accomplished by "twisting off" the throttle and simultaneously lowering the collective pitch control resulted in the rotor speed's exceeding the maximum power-off limit speed of 339 rpm. After "twisting off" the throttle, a momentary delay in lowering the collective pitch was necessary to prevent this overspeed condition. Entries into autorotation following an engine failure were simulated by using a rapid "throttle chop" followed by a 2-second delay before collective pitch control was lowered. Handling qualities during this transient maneuver will be discussed in Paragraph 2.6.4.6.

The contractor recommended that autorotational descents be accomplished above 60 KIAS and the UH-1B/540 rotor helicopter had satisfactory handling qualities in stabilized autorotation above 60 knots. Stabilized autorotational descents at 40 KIAS were difficult because a low-frequency large-amplitude yawing oscillation was experienced. Light-weight, low-altitude, full-down collective autorotational descents produced rotor speeds that were well above the minimum power-off rotor speed of 300 rpm. It was, however, necessary to apply collective pitch control to prevent a rotor overspeed condition when operating above the design gross weight of 6600 pounds.

The autorotational landing technique required for the UH-1B/540 helicopter was sufficiently different from that required for the standard UH-1B as to necessitate a pilot check-out. The cyclic flare prior to touchdown with the UH-1B/540 rotor helicopter had to be initiated at a higher altitude and held longer to produce the same effect as a flare with the standard UH-1B. At heavy gross weights, the cyclic flare did not produce an increase in rotor speed; therefore, it was necessary to maintain a rotor speed near the power-off limit of 339 rpm. At light gross weights, the flare produced approximately 10-rpm increase in rotor speed. The longitudinal control power was low but the pitching moments produced to level the helicopter prior to touchdown were satisfactory because the high rotor inertia permitted the pilot to apply collective pitch before the helicopter was level. After the helicopter was leveled, a steady application of collective pitch produced a smooth run-on landing. Touchdowns were made with a rotor speed as low as 235 rpm without encountering controllability problems. The 540 rotor at 249 rpm had the same energy, or stopping power, as the standard rotor at 324 rpm and the increased inertia was most noticeable after the helicopter was leveled and pitch was applied to slow the rate of sink.

Maximum skid height for hovering autorotations IGE was determined to be 5 feet for all gross weights. With the high-inertia 540 rotor system it was possible to hold the helicopter off the ground for approximately 5 seconds at the light gross weight and approximately 3 seconds at the heavy gross weights.

2.6 STABILITY AND CONTROL

2.6.1 Objective

The objectives of the stability and control tests were to obtain:

a. Qualitative control position data for trim speeds, in the range of 40 to 125 knots with forward, mid and aft C.G.'s.

b. Quantitative apparent speed stability for trim airspeeds in the range of 40 to 125 knots with forward, mid and aft C.G.'s.

c. Qualitative stick-free static longitudinal stability data (force gradient about a trim point) at various airspeeds and C.G.'s.

d. Quantitative stick-fixed static longitudinal stability data (position gradient about a trim point).

e. Qualitative and quantitative data on trim authority at high speeds with forward, mid and aft C.G.'s.

f. Qualitative and quantitative evaluation of handling qualities during entry into autorotation following a simulated engine failure.

g. Qualitative evaluation of control force harmony during maneuvering flight.

2.6.2 Method

Four flights were flown for stability and control data. Two flights each were flown at 9500 pounds gross weight with a forward C.G. and at 6500 pounds gross weight with an aft C.G. In addition, control positions were recorded for each stabilized trim point of the speed-power performance flight.

Static longitudinal stability data was obtained by varying airspeed with fore and aft cyclic control displacement with fixed collective control. The change in airspeed with change in stick position (position gradient) was a measure of stick-fixed stability. During this maneuver the stick force required to change airspeed was qualitatively evaluated by the pilot and was a measure of stick-free stability.

Handling qualities during entry into autorotation following a simulated engine failure were evaluated by recording control positions, roll and pitch angles, rotor rpm and pilot qualitative comments. Engine failure was simulated by retarding the throttle to the flight-idle stop, holding the collective fixed for 2 seconds, then lowering it within the next second. This time delay was used in compliance with Paragraph 3.5.5 of MIL-H-8501A (Reference 1) to simulate the delay in pilot reaction time following complete power failure. This maneuver is referred to in this report as a "throttle chop."

Control harmony, a measure of the relative magnitudes of directional, lateral, and longitudinal control forces, was qualitatively evaluated by performing nap-of-the-earth type maneuvering flight at various airspeeds, gross weights, and C.G.'s.

2.6.3 Results

Quantitative stability and control data is presented graphically in Figures 27 through 32.

Quantitative and qualitative data are discussed and analyzed in the following paragraphs.

2.6.4 Analysis

2.6.4.1 Control Positions

Longitudinal cyclic control position with an aft C.G. was uncomfortable at the light-weight, low-altitude, power-limit airspeeds. Extended operation at these conditions would be fatiguing to the pilot as he would not be able to rest his arm on his knee as is desired. This control position was evaluated by first adjusting the pilot seat to provide a comfortable collective control position, then determining if the longitudinal control position was satisfactory or unsatisfactory. This condition would be aggravated by the installation of a larger engine that would permit higher airspeeds in level flight.

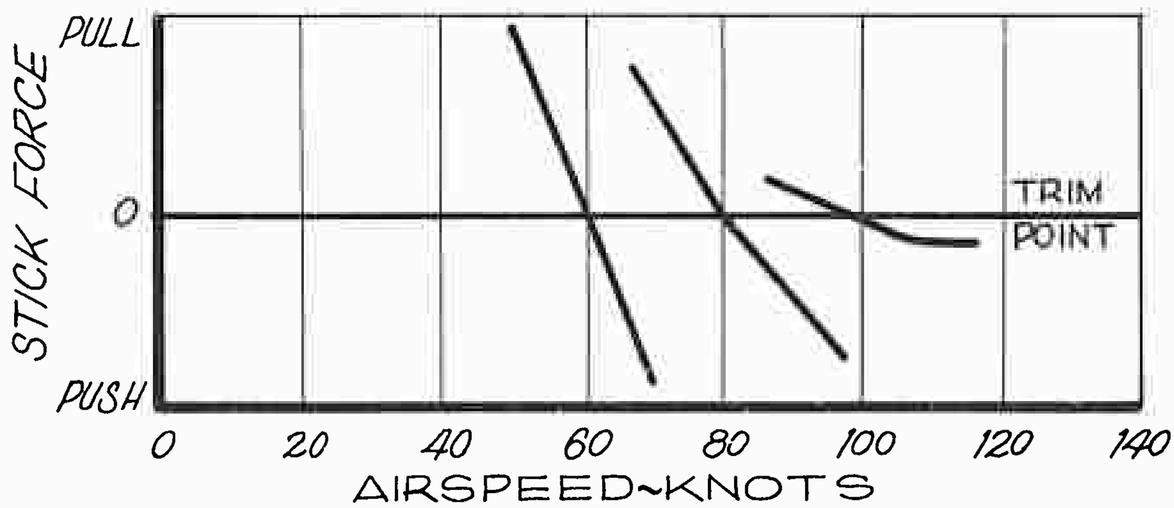
2.6.4.2 Speed Stability

The apparent speed stability (control position for trim airspeeds) was satisfactory with a forward C.G. (Figure 27). As the C.G. was moved aft the position gradient became more shallow as static stability decreased but was still within the requirements of Paragraph 3.2.10 of MIL-H-8501A (Reference 1). With an aft C.G., a 1-inch control displacement produced a 50-knot variation (75 knots to 125 knots) in airspeed. This, combined with the weak force gradient, made it difficult to trim the helicopter because the slight stick motion associated with pushing the trim button was enough to vary airspeed from the desired trim speed.

2.6.4.3 Stick-Free Static Longitudinal Stability

Stick-free static longitudinal stability (force gradient) was qualitatively evaluated as satisfactory up to an airspeed of 60 KCAS. Above this airspeed the force gradient became nonlinear and less force was required to increase airspeed than to decrease airspeed about a trim point. This nonlinear force gradient was caused by unbalanced forces greater than the spring in the trim force system. Forces such as stick weight, control rod and bell crank weights, and rubber boots that acted as springs all worked in the direction to move the stick forward and produce a neutral to negative force required to increase speed. The helicopter with an aft C.G. was trimmed at 110 KIAS. Speed was then increased without retrimming to 130 KIAS; and the stick when released fell forward, indicating that a negative stick force was required to increase airspeed. The stick force required to decrease airspeed about this trim point was greater than the force required to increase airspeed because the pilot was required to work against the normal spring force as well as the unbalanced forces described. Figure G presents graphically the trends of the UH-1B/540 rotor helicopter longitudinal force gradients.

FIG. G



It is desirable that longitudinal force gradients become steeper as maximum airspeeds are approached to prevent the pilot from inadvertently exceeding limit airspeeds and/or g loads. As shown in Figure G, the UH-1B/540 rotor helicopter longitudinal force gradients were exactly opposite. This condition plus the nonlinear force gradient and the relatively shallow position gradient combined to make it difficult to trim and fly with precision at the power-limit airspeed. The contractor's fix for this condition is discussed in Paragraph 1.3.2 of "Phase C Test Results," Appendix V.

2.6.4.4 Stick-Fixed Static Longitudinal Stability

Stick-fixed static longitudinal stability (Figures 28 and 29) was unsatisfactory above calibrated airspeeds of 100 knots. With a forward C.G. a 20-knot change in airspeed was obtained with only a .25-inch control displacement. An aft C.G. produced a highly unsatisfactory shallow position gradient about a trim speed of 109 KCAS. This meant that a change in airspeed of as much as 40 KCAS could be made without an apparent change in control position. The contractor's fix for this condition is discussed in Paragraph 1.3.2 of "Phase C Test Results," Appendix V.

2.6.4.5 Trim Characteristics

Adequate trim authority was available to trim the UH-1B/540 rotor helicopter at high speeds. As described in Paragraph 2.6.4.3 trim was difficult, however, because of the flat position gradient and weak, nonlinear force gradient. As noted in Paragraph 2.1.4.1, ...sufficient trim was available to trim control forces to zero during rearward flight. Testing of the prototype 540 rotor system revealed just the opposite. On the prototype 540 it was possible to trim in rearward flight, but insufficient trim authority was available to trim for high-speed forward flight. The contractor has shifted trim authority to the high-speed range and transferred the problem to the low-speed rearward flight range. Trim authority for the entire flight envelope, as recommended in USAAVNTA Report ATA-TR-64-2 (Reference c) and required by Paragraph 3.2.3 of MIL-H-8501A (Reference 1), is most desirable.

2.6.4.6 Handling Qualities

Handling qualities during entry into autorotation following a simulated engine failure were unsatisfactory. Throttle chops executed at high speed resulted in the normal yawing to the left followed by pitching nose down and rolling to the left. Throttle chops executed with fixed cyclic control resulted in a 10-degree nose-down, 30-degree left roll attitude within 2 seconds.

To hold a constant attitude following a simulated power failure, a right lateral and an aft longitudinal cyclic change was required. The forward stick position associated with high speed made it difficult for the pilot to apply the force required to make a right lateral trim change. In addition, the high rolling angular acceleration to the left caused the pilot to over-correct to the right, thus increasing the lateral force required. The undesirable cyclic trim changes of the production 540 rotor system contrasted with the excellent autorotation entry characteristics of the prototype 540 rotor system (Reference c).

Rotor rpm decay during entry into autorotation was surprisingly high for a rotor with as much inertia as the 540 rotor system. A throttle chop initiated at 324 rpm, 131 KCAS, followed immediately by a flare to 60 KCAS resulted in a maximum rpm decay rate of 30 rpm per second 1.0 second after the throttle was chopped (Figure 30). During the simulated engine failure tests at high speed, it was necessary to lower the collective rapidly, following a 2-second delay, to prevent excessive rpm droop. This was exactly the opposite type of reaction required during entry into practice autorotations at lower speeds, when it was necessary to exercise caution to prevent an overspeed condition (Paragraph 2.5.4.2). A pilot accustomed to lowering the collective slowly for practice autorotations would probably not lower the collective rapidly enough in case of an engine failure to prevent excessive rpm droop. Entry into autorotation holding a constant attitude (i.e., without flaring) produced an rpm decay rate of approximately 35 to 38 rpm per second within 0.5 to 1.0 seconds after the throttle was chopped (Figures 31 and 32).

Controllability problems due to low rotor speed were not encountered even though all throttle chops resulted in rotor rpm droop below the minimum power-off limit of 300 rpm. For this reason, consideration should be given to establishing a lower power-off rotor limit.

Rotor rpm buildup was satisfactory at the heavier gross weights (Figure 33) but was unsatisfactory at the lighter gross weights. At 6600 pounds gross weight the time required to regain minimum rotor speed after placing the collective in the full-down position was 5.0 seconds. In this case, the low rpm warnings were operating for a period of 8.0 seconds and this was disconcerting to the pilot.

Control of rotor rpm during entry into autorotations was difficult because large collective control inputs were required to stop the rate of rpm decay and to check the rotor acceleration during rpm buildup. Control of the 540 rotor, especially at high gross weights, was sufficiently different from control of the standard UH-1B as to require a pilot check-out.

2.6.4.7 Control Force Harmony

Control force harmony during maneuvering flight was acceptable but could be improved. The force trim system of the UH-1B/540 rotor helicopter as tested during Phase B had a longitudinal-lateral force ratio of 1:1. The unbalanced forces, described in Paragraph 2.6.4.3, that created a weak longitudinal force gradient at high speeds also affected this marginal force ratio. In addition, these forces adversely affected control harmony since the relatively high lateral force requirements masked the weaker longitudinal forces during nap-of-the-earth flight.

Step-type cyclic control inputs required to perform nap-of-the-earth-type flight produced 3 to 4 cycles of high-amplitude vibrations followed by 10 to 12 cycles of low-amplitude, lightly damped residual vibration. This residual vibration was annoying and produced a condition that caused the helicopter to be in a continual state of vibration during maneuvering flight. This would reduce the pilot's tactical effectiveness. The normal g loads experienced during maneuvering flight, plus the vibration, caused the collective pitch control to slip unless excessive collective friction was applied. An acceptable collective control force of approximately 10 pounds was satisfactory for cruise flight but unsatisfactory for maneuvering flight. During maneuvering flight, a collective control friction force sufficient to prevent slippage of the collective was fatiguing to the pilot and a collective friction force less than 12 pounds increased pilot's workload to maintain a proper power setting. This portion of control harmony, therefore, is a matter of pilot preference and is determined by the type of mission flown. (See "Phase C Test Results," Appendix V).

2.7 AIRSPEED CALIBRATION

2.7.1 Objective

The ship's airspeed system was calibrated to determine the position error in indicated airspeeds caused by the pitot-static tube location.

2.7.2 Method

The position error of the ship's airspeed system was determined by using the "trailing bomb" method in level and climbing flight. The "trailing bomb" incorporated static and dynamic ports located so as to produce airspeed indications with zero position error. This device was towed below the test helicopter. The difference in instrument-corrected indicated airspeeds determined the ship's position error for each flight condition.

2.7.3 Results

Results of the airspeed calibration are presented graphically in Figure 33.

2.7.4 Analysis

The test helicopter was not equipped with a special airspeed boom; therefore, all comments in this report pertain to the ship's standard airspeed system that is described in Appendix IV. The curves used to present level and climbing flight position error were obtained from an airspeed calibration conducted by the contractor. These curves agreed favorably with the Phase B airspeed calibration data points. Additional data that is required to establish position error in autorotational flight will be obtained during Phase D.

The airspeed system had a negative position error at all speeds above 40 KIAS with a maximum correction of 4 knots at 140 KIAS. With a negative position error, the helicopter speed was actually slower than indicated.

In climbing flight the airspeed system had a positive position error that varied from 3 to 6 knots within the airspeed range for maximum rate of climb (55 to 60 knots). This variation in position error was undesirable because it was difficult to fly accurately the optimum climb speed schedule.

2.8 ROTOR BLADE TRACKING

2.8.1 Objective

The objective of this test was to learn if special techniques were required as compared to rotor tracking in a standard UH-1B helicopter.

2.8.2 Method

Prior to the initial Phase B test flight the contractor tracked the main rotor to eliminate as much as possible the rotor induced vibration. USAAVNTA personnel observed this procedure.

2.8.3 Results

Approximately four days and numerous short test flights were required before the contractor was satisfied with the vibration levels in all flight regimes. Standard blade tracking procedures were used (Reference f, Paragraph 2.h(2)).

The vibration tests revealed that the 1-per-rev vibrations increased slightly, the 2-per-rev vibrations decreased significantly and the 4-per-rev vibrations were similar to those of a standard UH-1B at similar loading conditions. As a general trend, the 1-per-rev and 2-per-rev vibration levels increased and the 4-per-rev vibration level decreased as the C.G. was moved aft. The 1-per-rev vibration level remained unchanged with increasing gross weight, whereas the 2-per-rev and 4-per-rev vibration levels at the pilot and copilot stations increased with increasing gross weight.

As discussed under climb performance (Paragraph 2.3.4.4), the 1-per-rev frequency was the predominant vibration during climb and partial power descent. This vibration was encountered under a steady-state (climbing or descending) flight condition but was random in nature. In partial power descents an airspeed and power setting could be selected that would produce a 1-per-rev vibration that was no longer random but continuous in nature. This vibration was very disturbing and should be improved. The cause of this vibration, as discussed in Paragraph 2.3.4.4, was attributed to the fact that the blades were out of tolerance in profile contour at the inboard sections. As manufacturing methods improve, this 1-per-rev vibration characteristic probably also will be improved.

2.8.4 Analysis

The excessive amount of time required to track the 540 rotor system was attributed to the fact that the blade profile was out of production tolerances at the inboard section. Rotor blade tracking is also discussed in "Phase C Test Results," Appendix V.

2.9 CONTROL RIGGING CHECK

2.9.1 Objective

Prior to the initial Phase B test flight an inspection was made to determine if the test helicopter control system was rigged as shown on the contractor's Production Rigging Drawing 204-401-006 (Reference f).

2.9.2 Method

The control rigging check was made with the helicopter level and with hydraulic boost on. The following measurements were taken:

a. Mast angle (longitudinal and lateral)

b. Cyclic stick position

c. Collective stick position

d. Pedal position

e. Swashplate position

f. Main rotor blade angle

g. Tail rotor blade angle

h. Main rotor blade flapping angle

i. Tail rotor blade flapping angle

j. Stabilizer low flapping angle

k. Synchronized elevator position

2.9.3 Results

The test helicopter's control system was rigged as shown on the contractor's Production Rigging Drawing 204-401-006 (Reference f, Paragraph 2.h(1)).

2.9.4 Analysis

After completion of Phase B testing a change was made to the sychronized elevator position. This is discussed in "Phase C Test Results," Appendix V.

2.10 WEIGHT AND BALANCE

2.10.1 Objective

The objective of this test was to weigh the test helicopter prior to the first Phase B test flight to determine aircraft basic weight and C.G.

2.10.2 Method

The test helicopter was leveled and weighed in a closed hangar using the contractor's weight and balance facilities.

2.10.3 Results

The basic weight of 5176 pounds included test instrumentation, full oil, trapped fuel and three crew member seats.

2.10.4 Analysis

No attempt was made to estimate the basic weight of the tested first production UH-1B/540 rotor helicopter from this weight and balance data because of the unknown weight of the test instrumentation. The contractor, however, has provided, in Reference f, the following weight estimates:

Component	Weight Increase From Standard UH-1B lb
Hub	112
Blades	60
Rotating Controls	32
Elevator Controls	5
Combered Fin Tail Boom	4
Dual Boost	50
TOTAL	263 lb

This weight breakdown does not include the increase in weight due to the installation of the UH-1D elevator (See Appendix IV). The contractor has estimated the basic weight of UH-1B/540 rotor helicopter, S/N 64-14101 (the first production UH-1B/540), to be 4842 pounds as submitted in the Model Specification 204-947-125 (Reference o).

SECTION 3 APPENDICES

APPENDIX I - Test Data

FIGURE No. 1
CLIMB PERFORMANCE
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM
STANDARD DAY
GROSS WEIGHT ~ 7660 LB
ROTOR SPEED ~ 324.0 RPM
C.G. ~ STATION 131.0 (MID)
T53-L-11 ENGINE

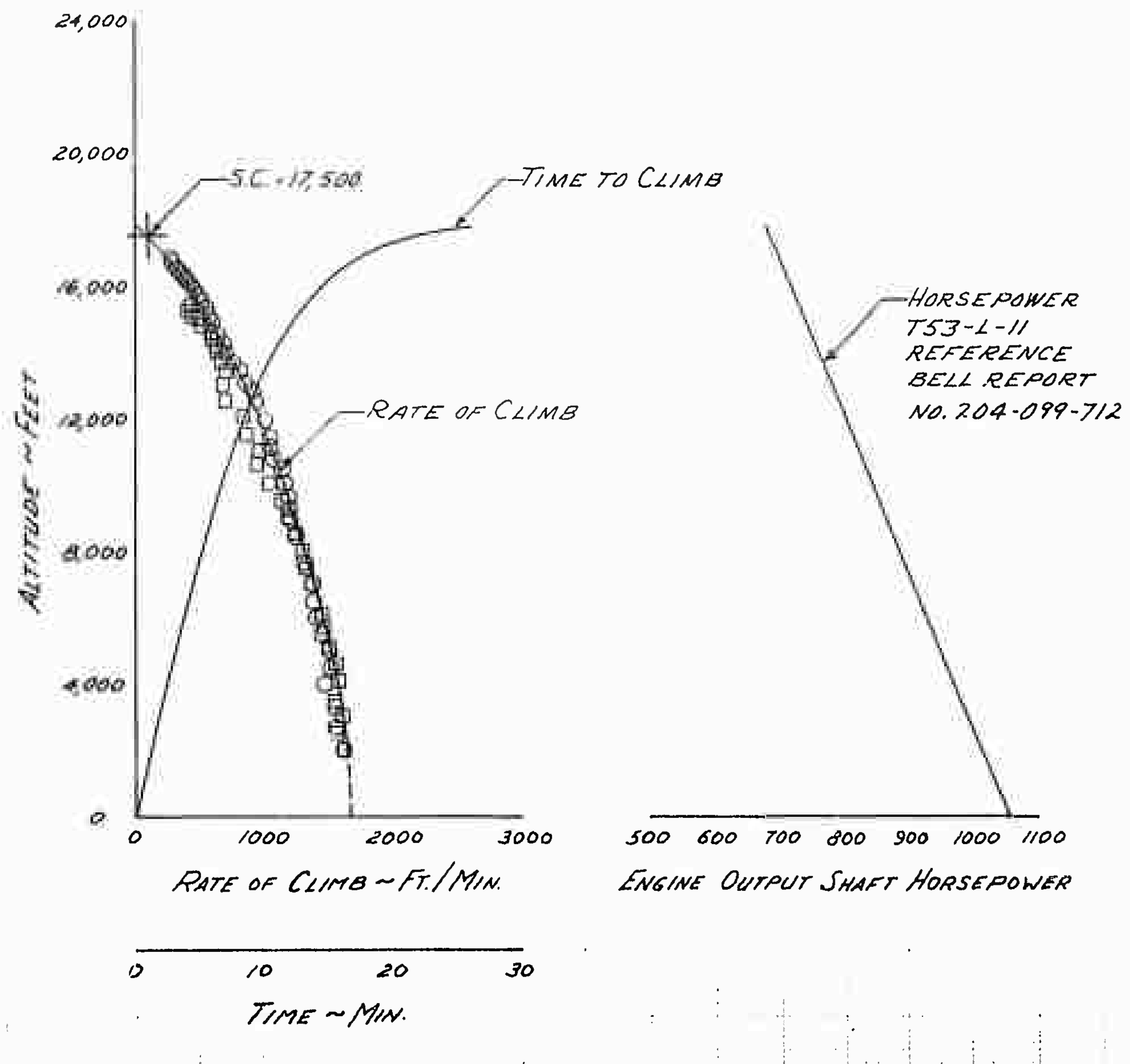


FIGURE No. 1 (CONT'D)

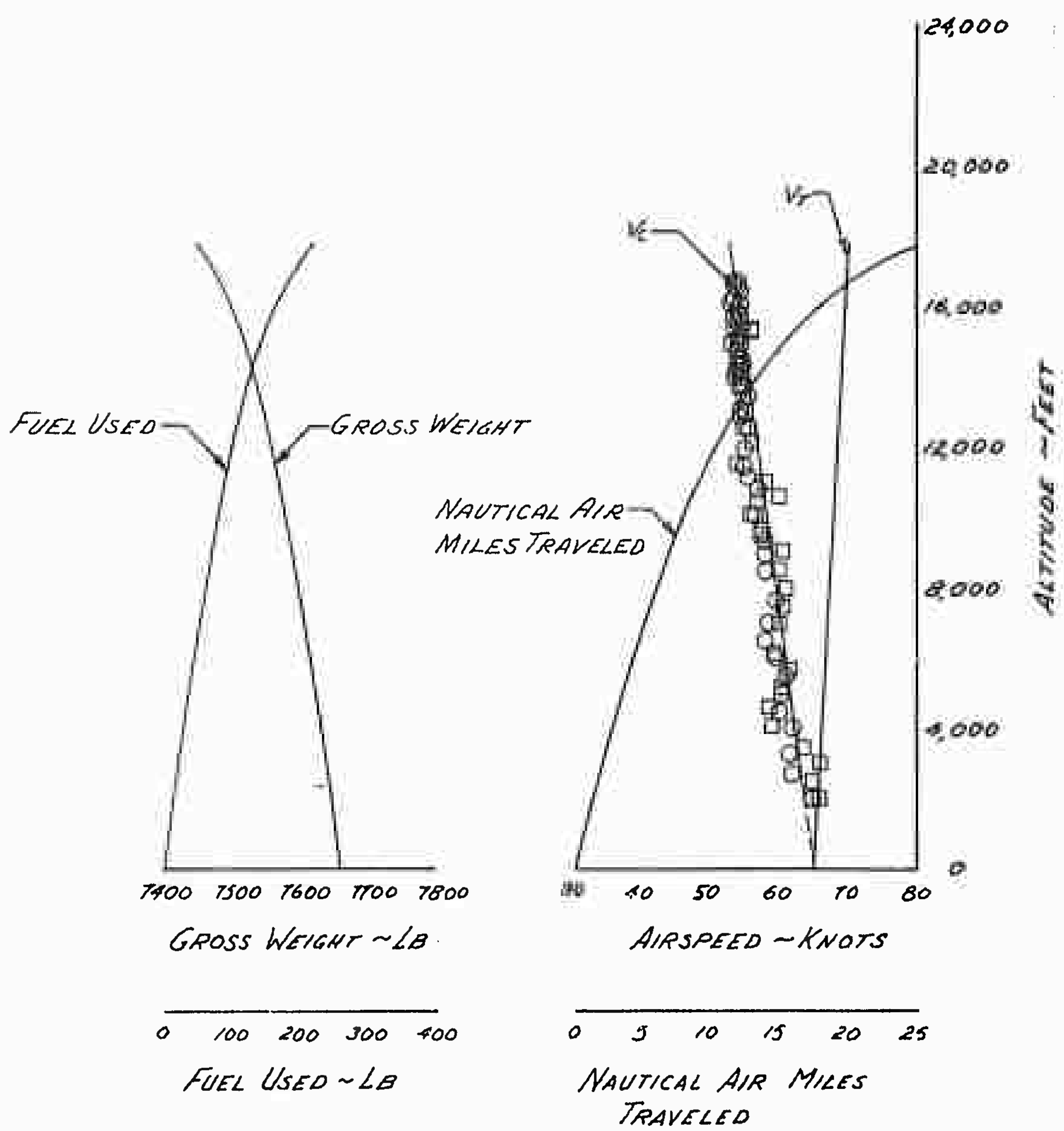


FIGURE NO. 2
LEVEL FLIGHT SUMMARY
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM
T53-L-11 ENGINE

SYM.	RPM	C.G.
○	324.0	131.0 (MID)
□	324.0	136.0 (AFT)
△	324.0	126.0 (FWD)

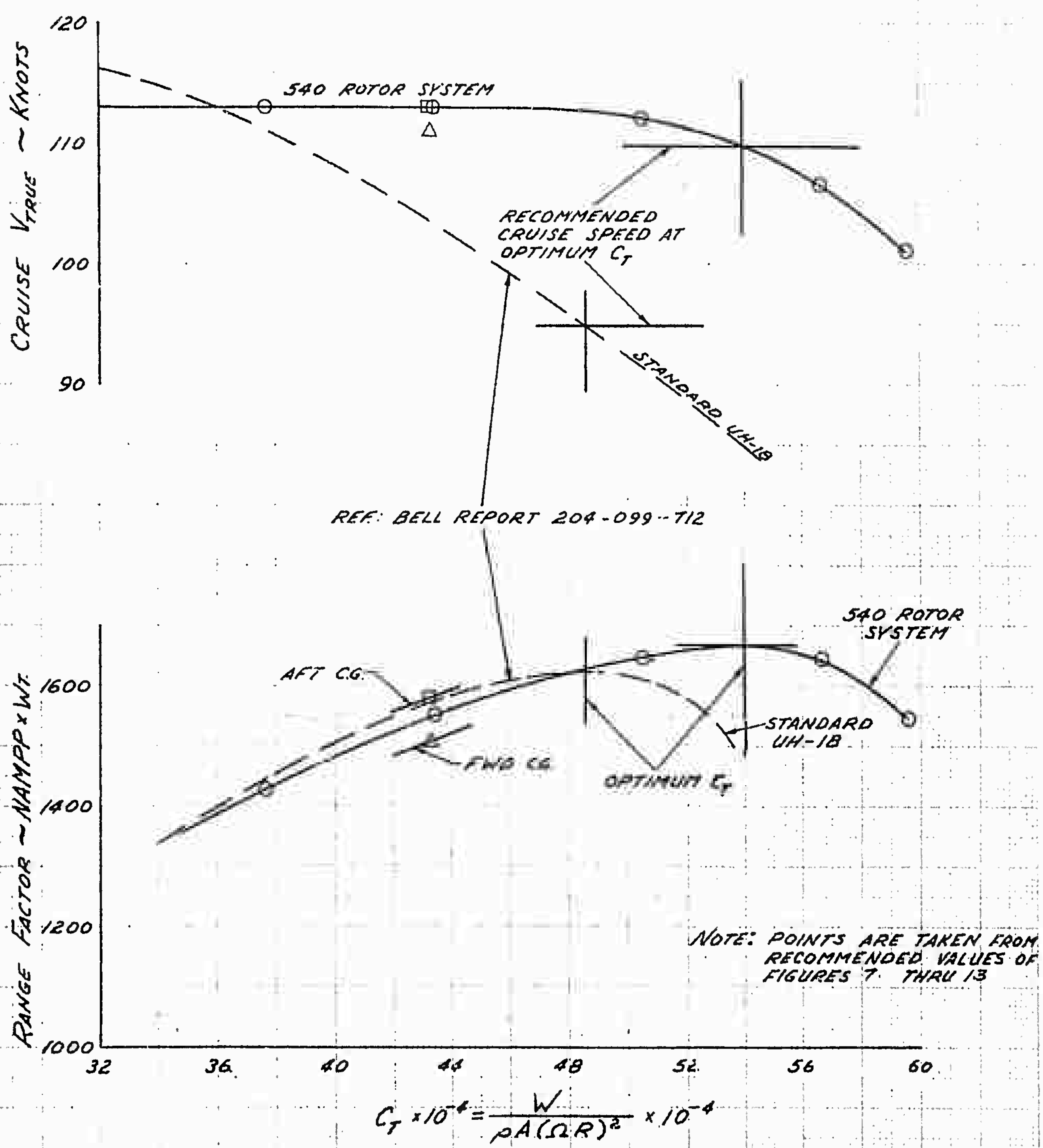


FIGURE NO. 3

NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-1B USA S/N 63-8684

BELL MODEL 540 ROTOR SYSTEM

T53-L-11 ENGINE

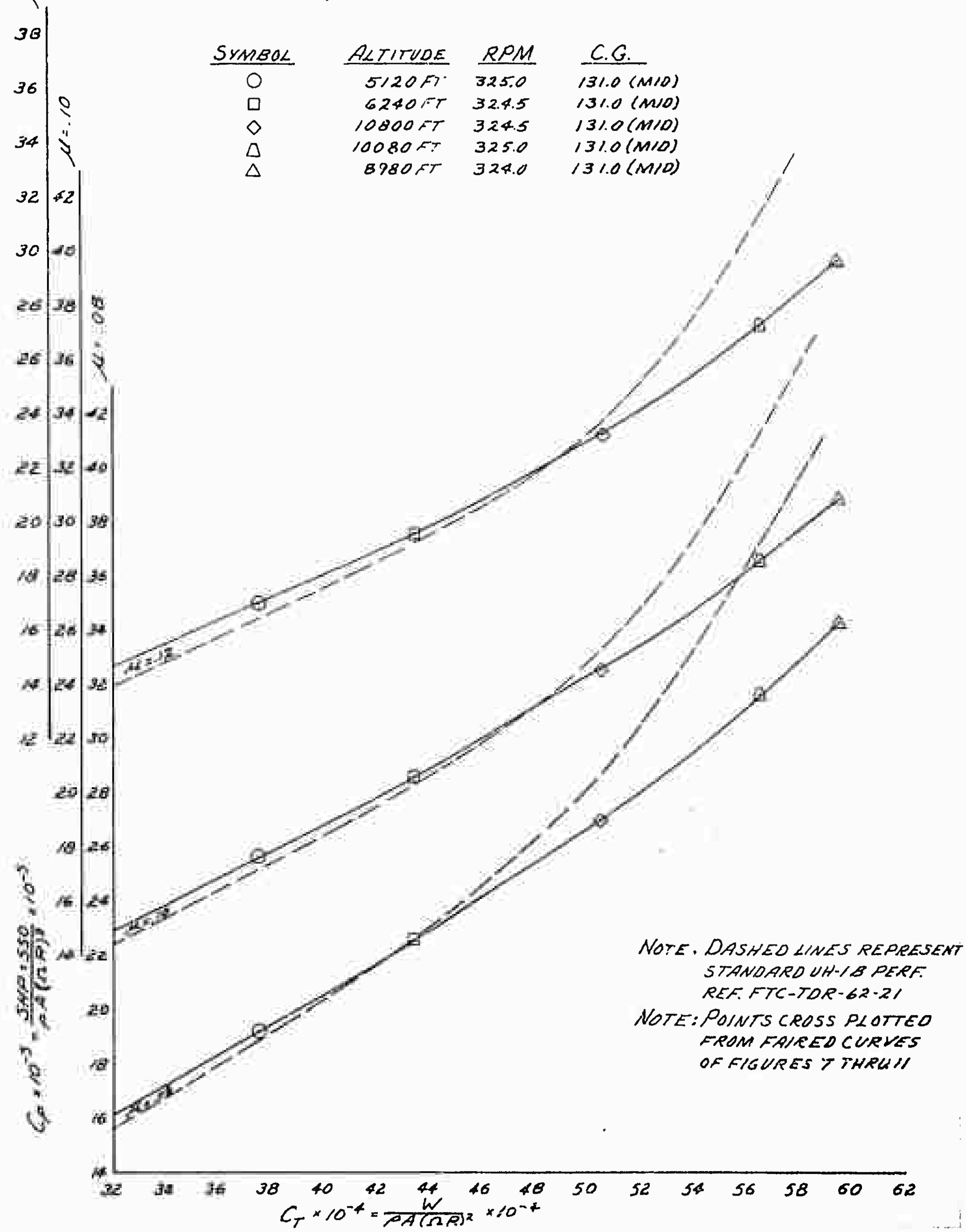


FIGURE NO. 4
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM
T53-L-11 ENGINE

SYMBOL	ALTITUDE	RPM	C.G.
○	5120 FT	325.0	131.0 (MID)
□	6240 FT	324.5	131.0 (MID)
◇	10800 FT	324.0	131.0 (MID)
⊿	10080 FT	325.0	131.0 (MID)
△	8980 FT	324.0	1310 (MID)

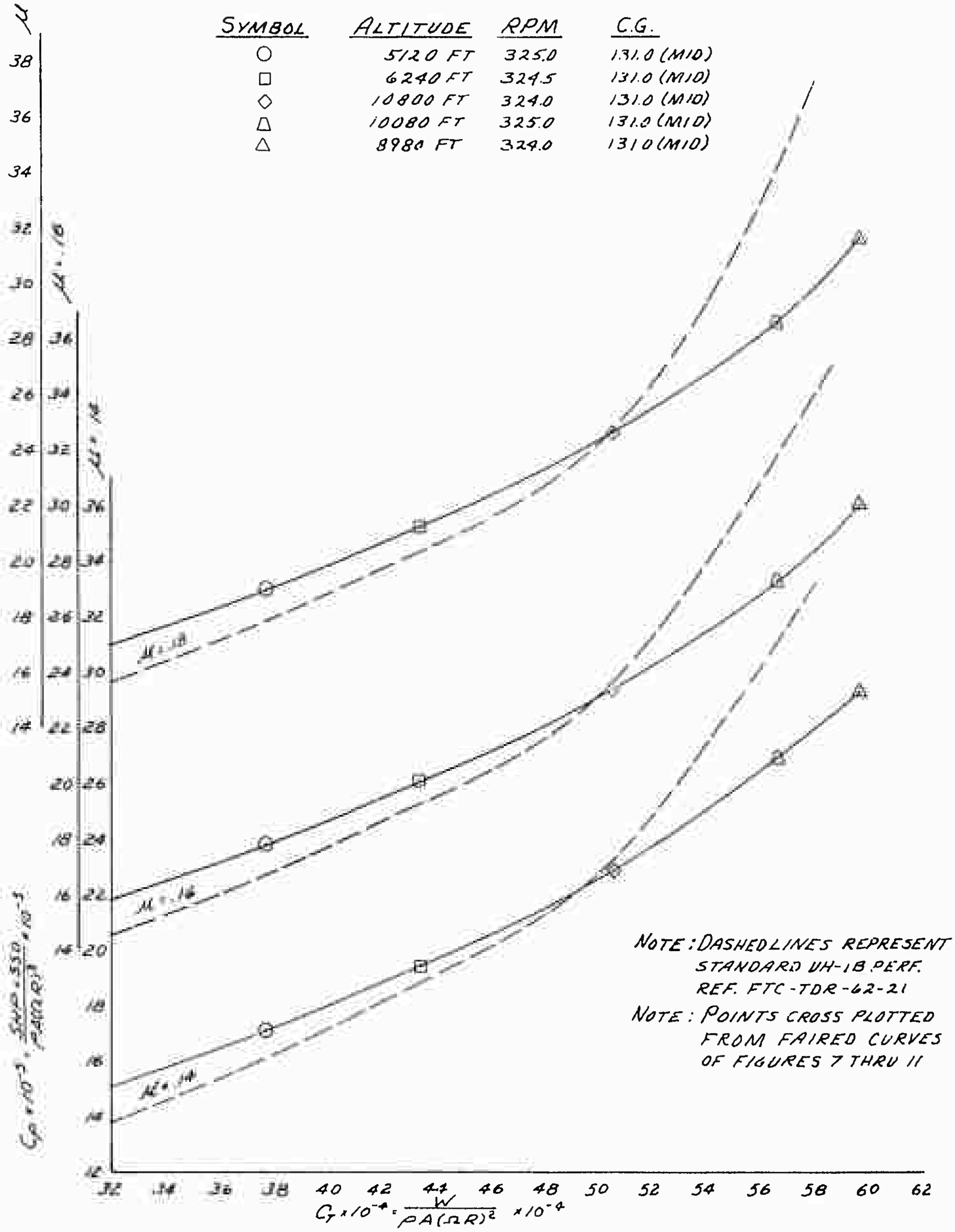


FIGURE No. 5
NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM
T53-L-11 ENGINE

SYMBOL	ALTITUDE	RPM	C.G.
○	5120 FT	325.0	131.0 (MID)
□	6240 FT	324.5	131.0 (MID)
◇	10800 FT	324.5	131.0 (MID)
△	10080 FT	325.0	131.0 (MID)
△	8980 FT	324.0	131.0 (MID)

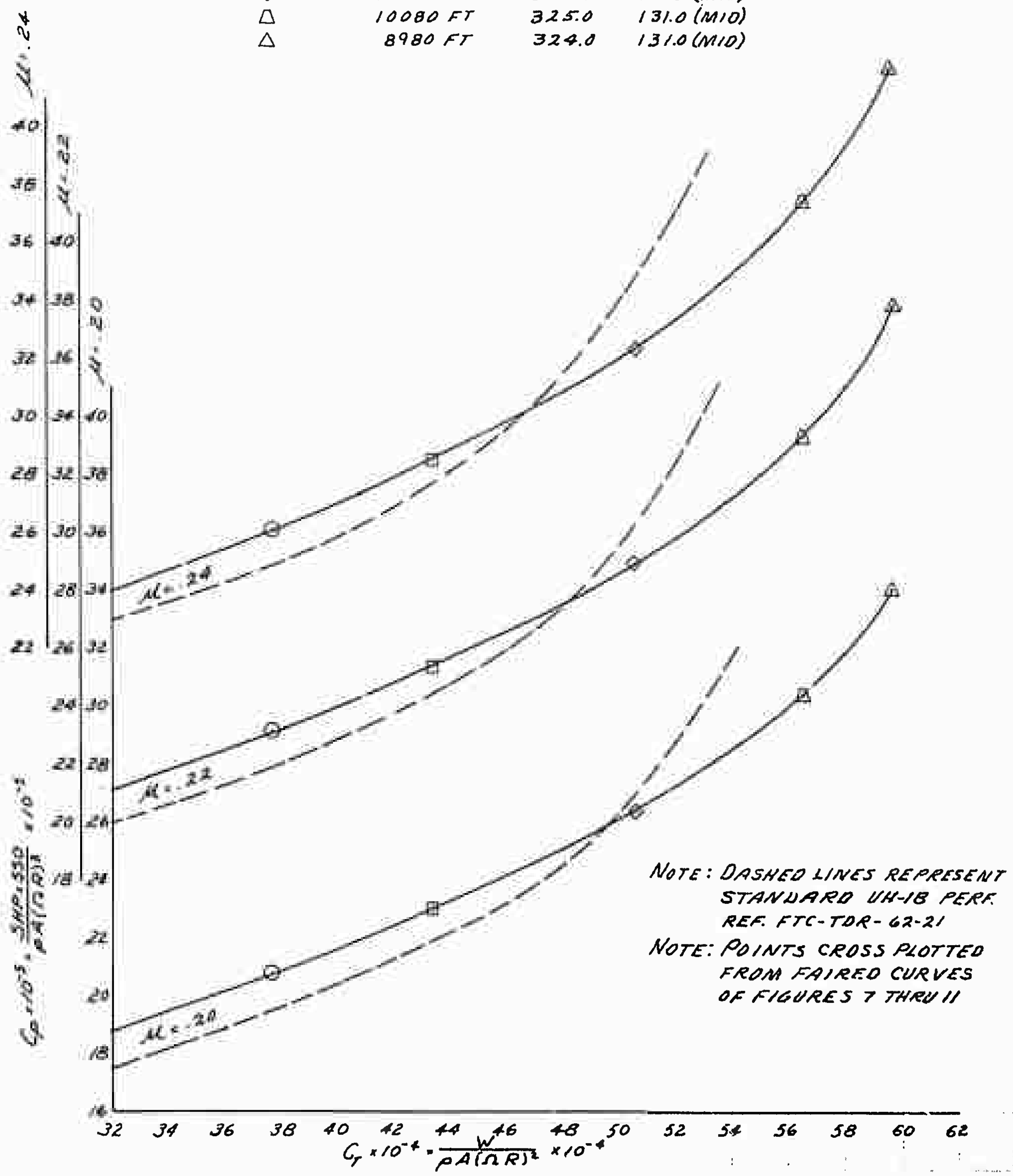


FIGURE NO. 6

NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE

UH-1B USA S/N 63-8684

BELL MODEL 540 ROTOR SYSTEM

T53-L-11 ENGINE

SYMBOL	ALTITUDE	RPM	C.G.
○	5120 FT	325.0	131.0 (MID)
□	6290 FT	324.5	131.0 (MID)
◇	10800 FT	324.5	131.0 (MID)
⏢	10080 FT	325.0	131.0 (MID)
△	8930 FT	324.0	131.0 (MID)

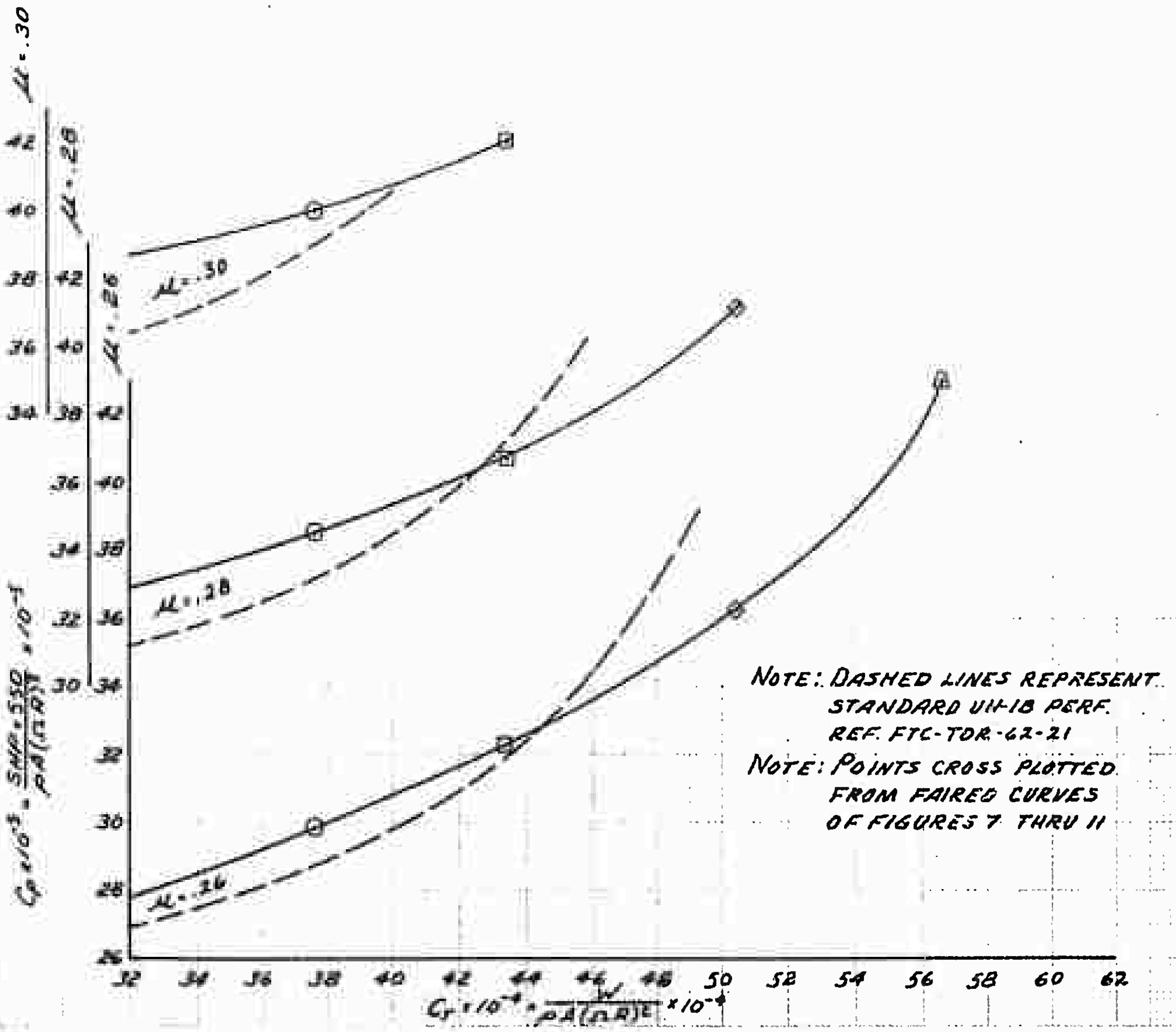
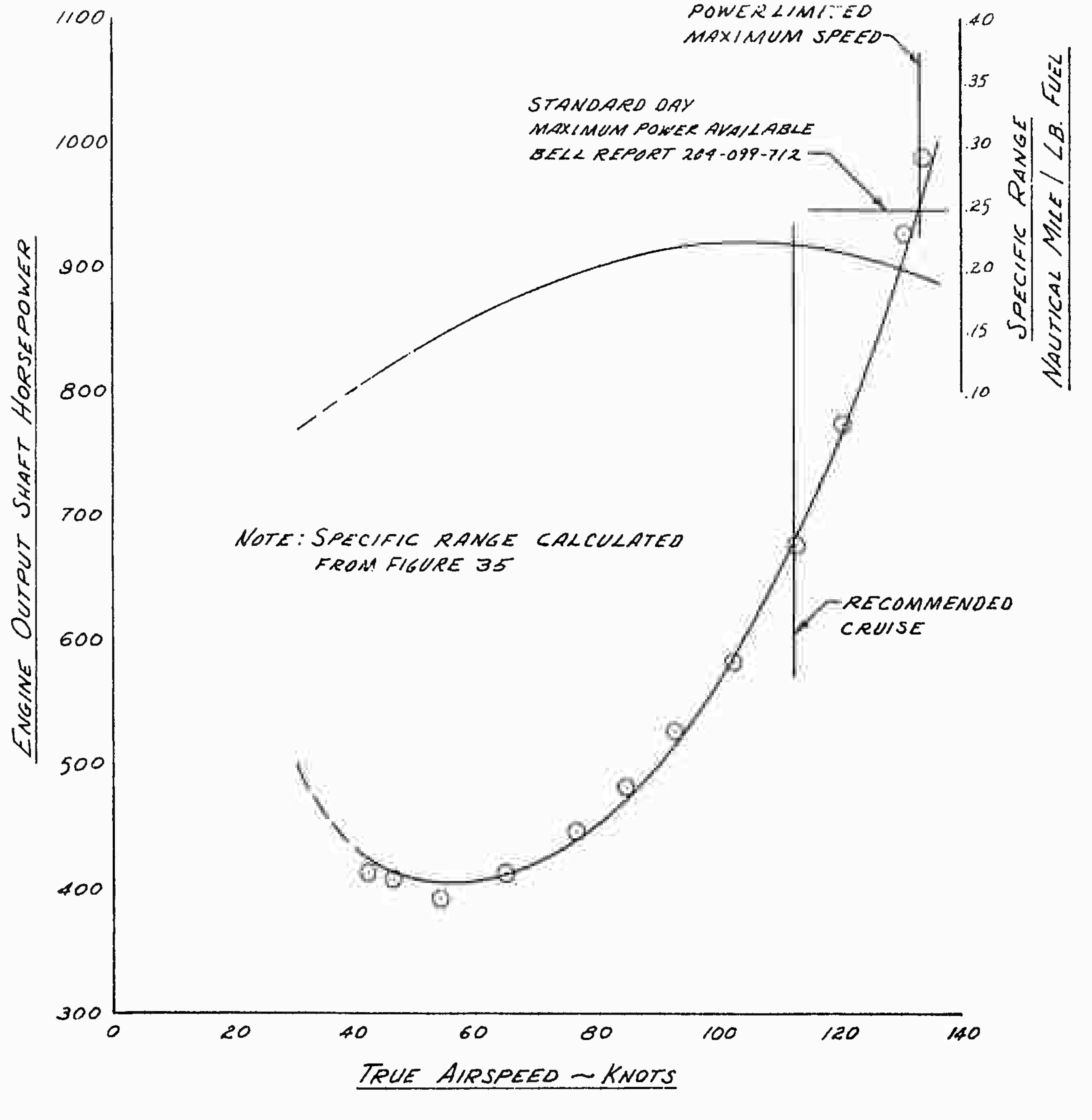


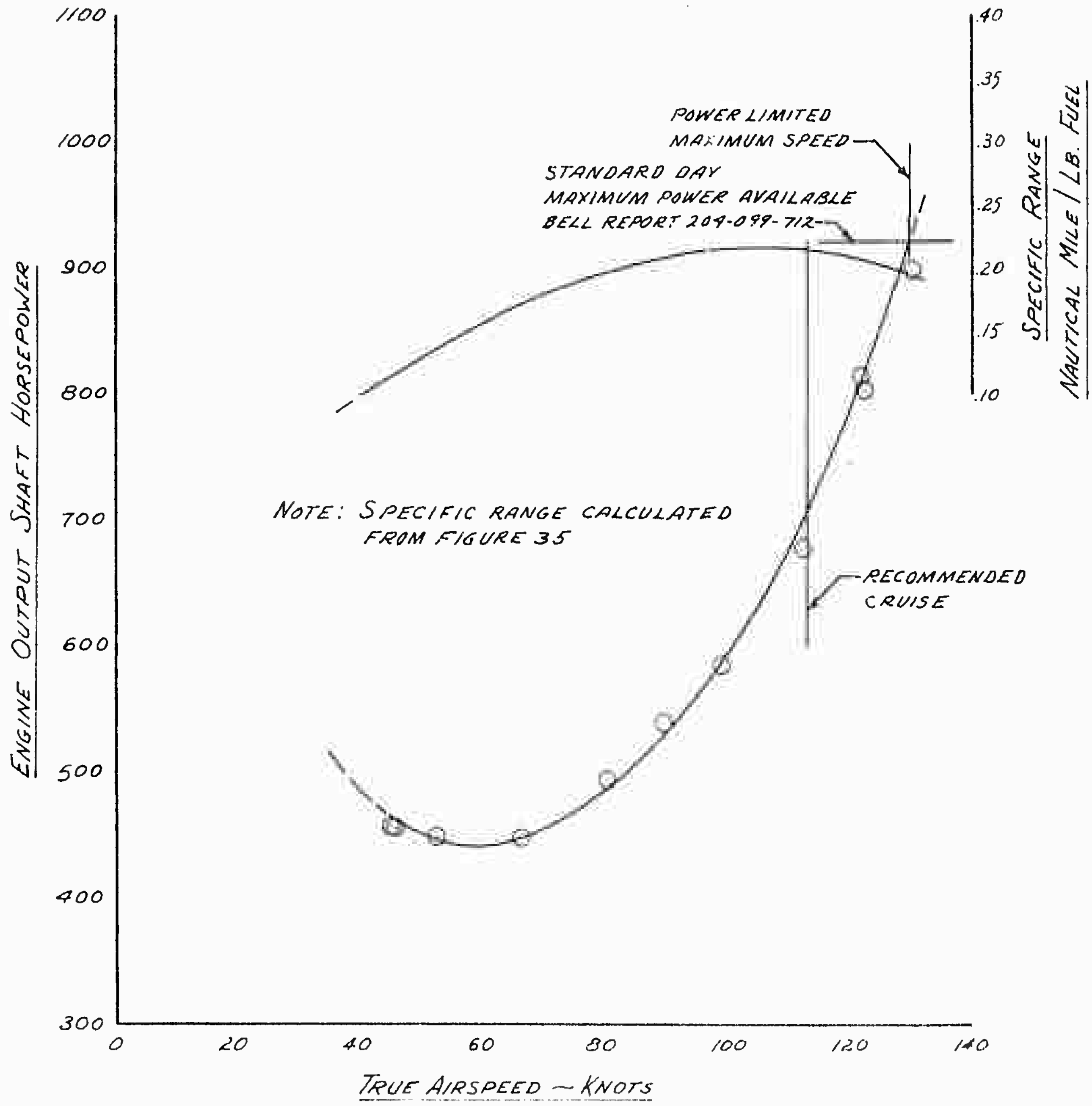
FIGURE NO. 7
LEVEL FLIGHT PERFORMANCE
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM
T53-L-11 ENGINE

GROSS WEIGHT ~ 6555 LB
DENSITY ALTITUDE ~ 5120 FT.
ROTOR SPEED ~ 325.0 RPM
C.G. ~ STATION 131.0 (MID)
C_T ~ .003766



FLIGHT No. 8
LEVEL FLIGHT PERFORMANCE
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM
T53-L-11 ENGINE

GROSS WEIGHT ~ 7270 LB
DENSITY ALTITUDE ~ 6240 FT.
ROTOR SPEED ~ 324.5 RPM
C.G. ~ STATION 131.0 (MID)
C_T ~ .004337



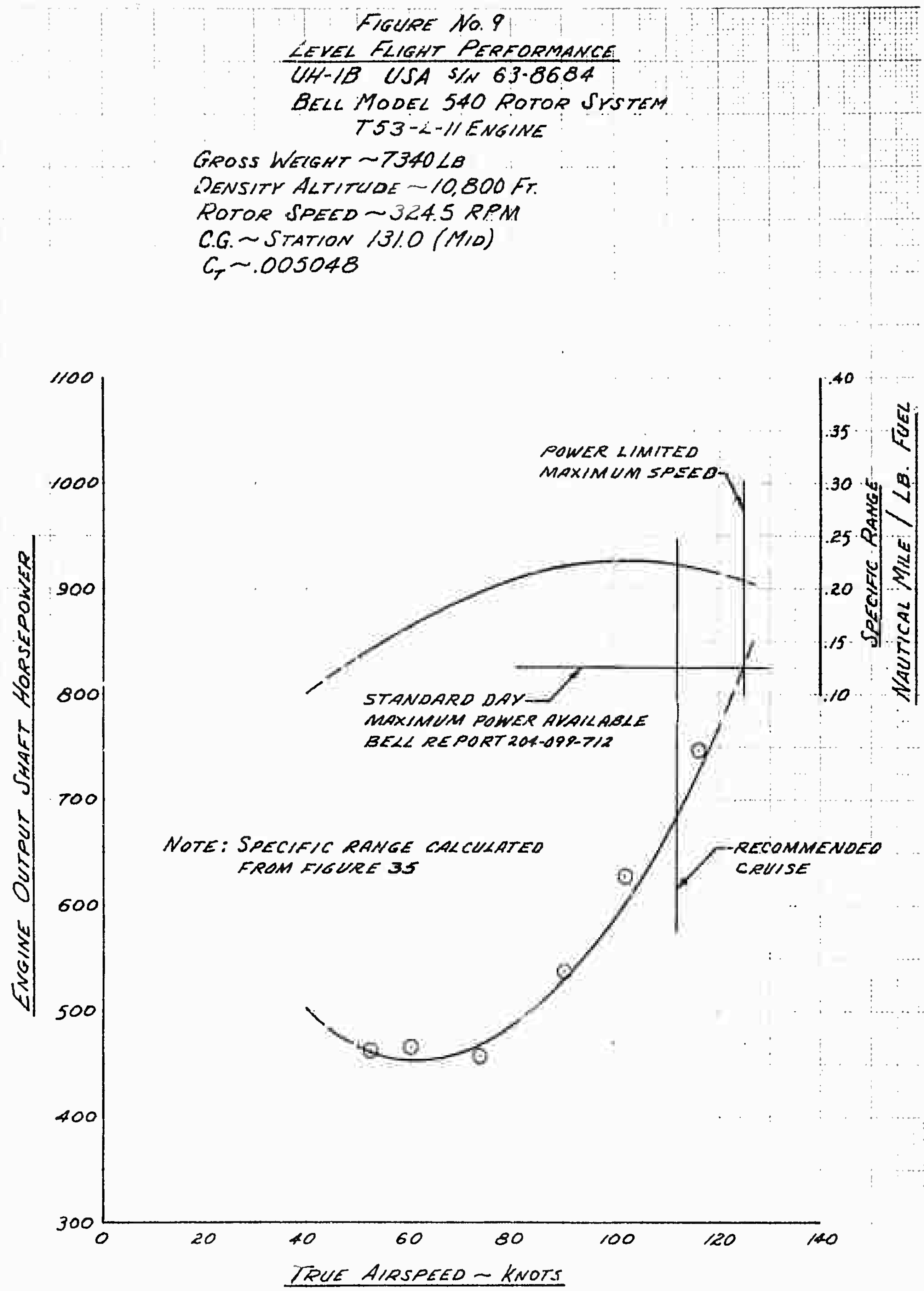


FIGURE No. 9
LEVEL FLIGHT PERFORMANCE
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM
T53-L-11 ENGINE
GROSS WEIGHT ~ 7340 LB
DENSITY ALTITUDE ~ 10,800 FT.
ROTOR SPEED ~ 324.5 RPM
C.G. ~ STATION 131.0 (MID)
C_T ~ .005048
POWER LIMITED MAXIMUM SPEED
STANDARD DAY MAXIMUM POWER AVAILABLE BELL REPORT 204-099-712
NOTE: SPECIFIC RANGE CALCULATED FROM FIGURE 35
RECOMMENDED CRUISE
ENGINE OUTPUT SHAFT HORSEPOWER
SPECIFIC RANGE NAUTICAL MILE / LB. FUEL
TRUE AIRSPEED ~ KNOTS
1100
1000
900
800
700
600
500
400
300
.40
.35
.30
.25
.20
.15
.10
0
20
40
60
80
100
120
140

FIGURE No. 10
LEVEL FLIGHT PERFORMANCE
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM
T53-L-11 ENGINE

GROSS WEIGHT ~ 8420 LB.
DENSITY ALTITUDE ~ 10,080 FT.
ROTOR SPEED ~ 325.0 RPM
C.G. ~ STATION 131.0 (MID)
C_T ~ .005664

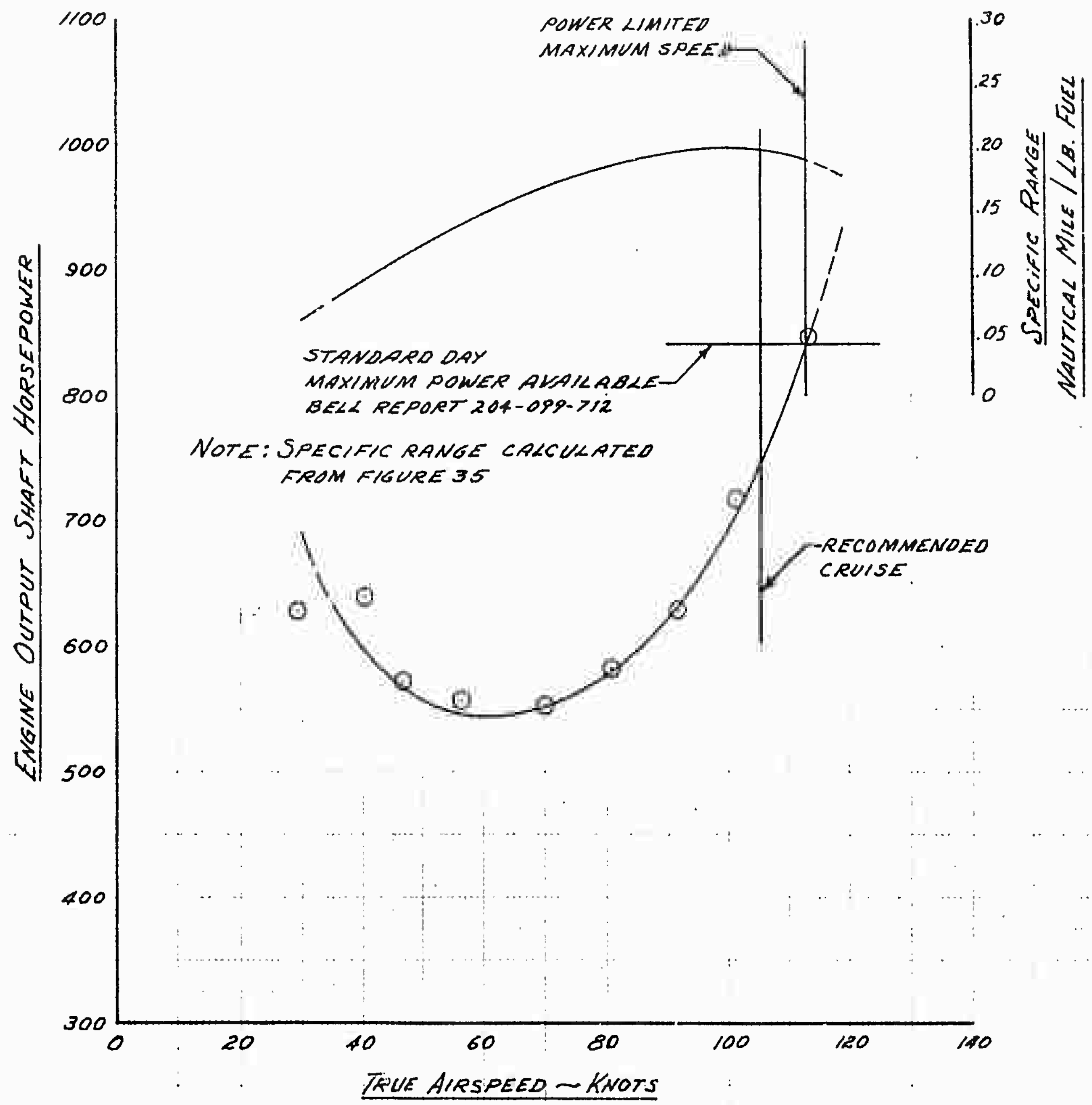


FIGURE No. 11
LEVEL FLIGHT PERFORMANCE
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM
T53-L-11 ENGINE

GROSS WEIGHT ~ 9160 LB
DENSITY ALTITUDE ~ 8980 FT.
ROTOR SPEED ~ 324.0 RPM
C.G. ~ STATION 131.0 (MID)
C_T ~ .005960

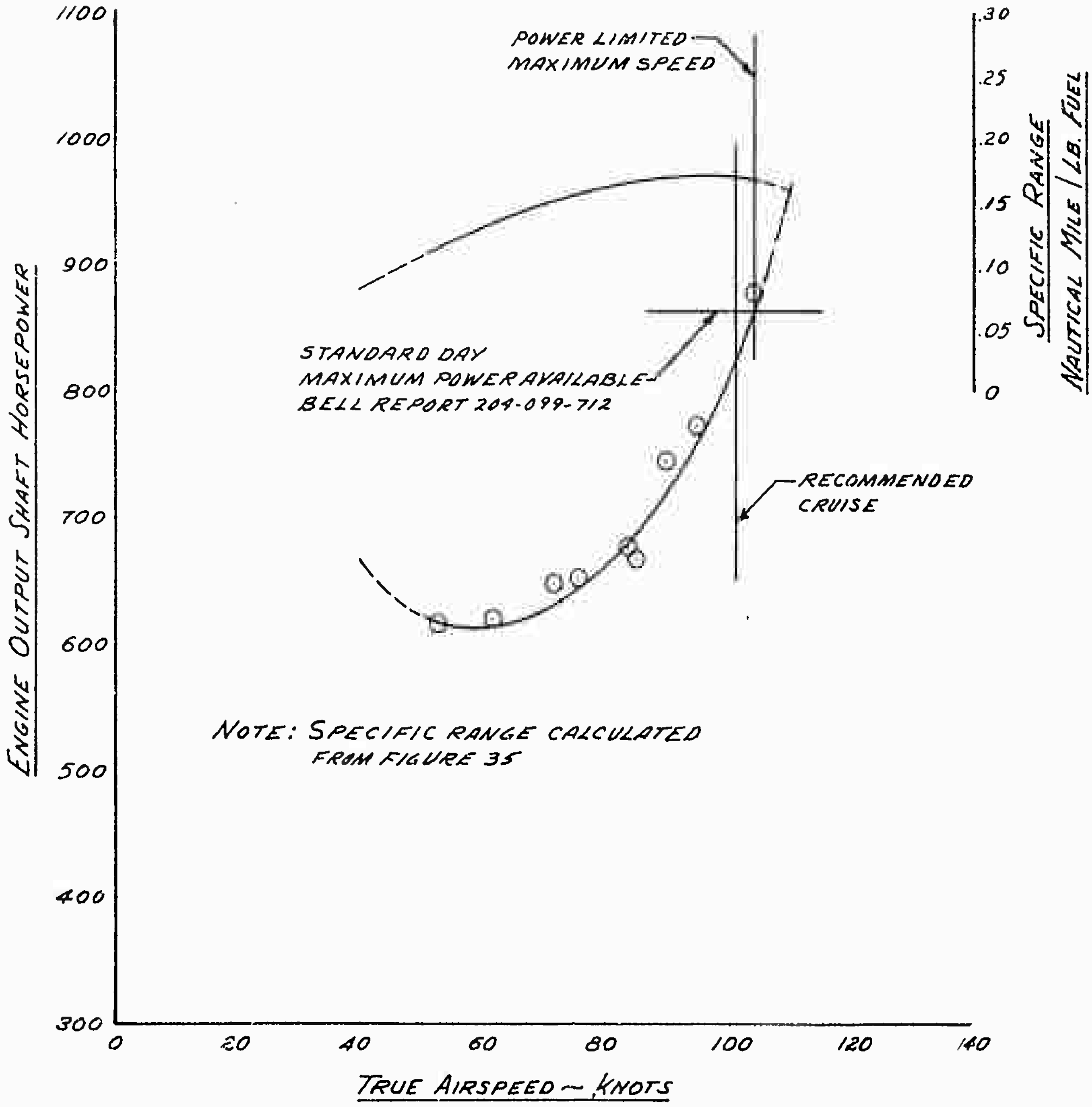


FIGURE NO. 12
LEVEL FLIGHT PERFORMANCE
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM
T53-L-11 ENGINE

GROSS WEIGHT ~ 7480 LB
DENSITY ALTITUDE ~ 5490 FT.
ROTOR SPEED ~ 325.5 RPM
C.G. ~ STATION 126.0 (FWD)
C_T ~ .004333

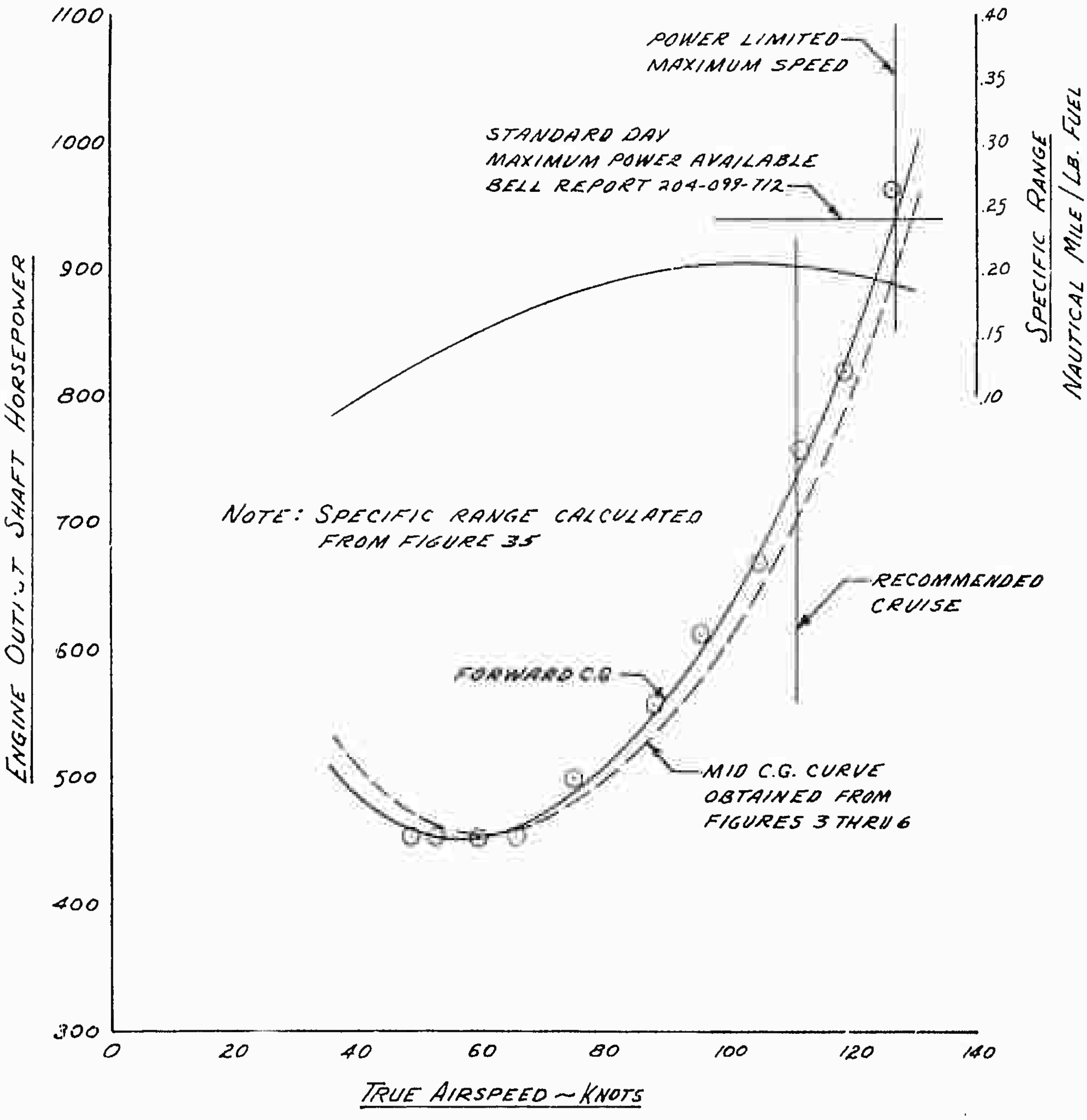


FIGURE NO. 13
LEVEL FLIGHT PERFORMANCE
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM
T53-L-11 ENGINE

GROSS WEIGHT ~ 7490 LB.
DENSITY ALTITUDE ~ 5140 FT.
ROTOR SPEED ~ 325.0 RPM
C.G. ~ STATION 136.0 (AFT)
C_T ~ .004311

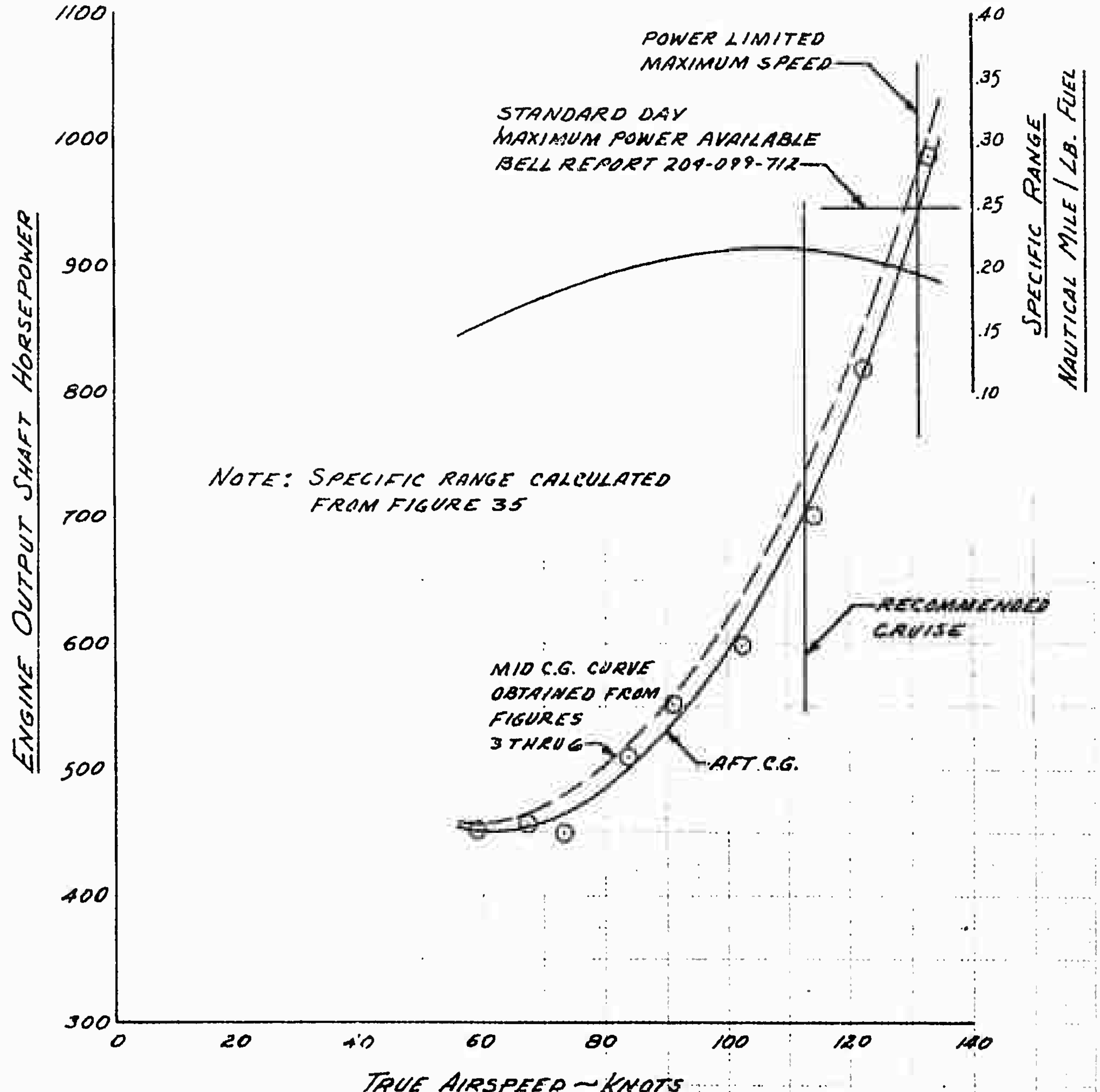


FIGURE No. 14
VIBRATION CHARACTERISTICS
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 7480 LB
DENSITY ALTITUDE ~ 5490 FT.
ROTOR SPEED ~ 325.5 RPM
C.G. ~ STATION 126.0 (FWD)

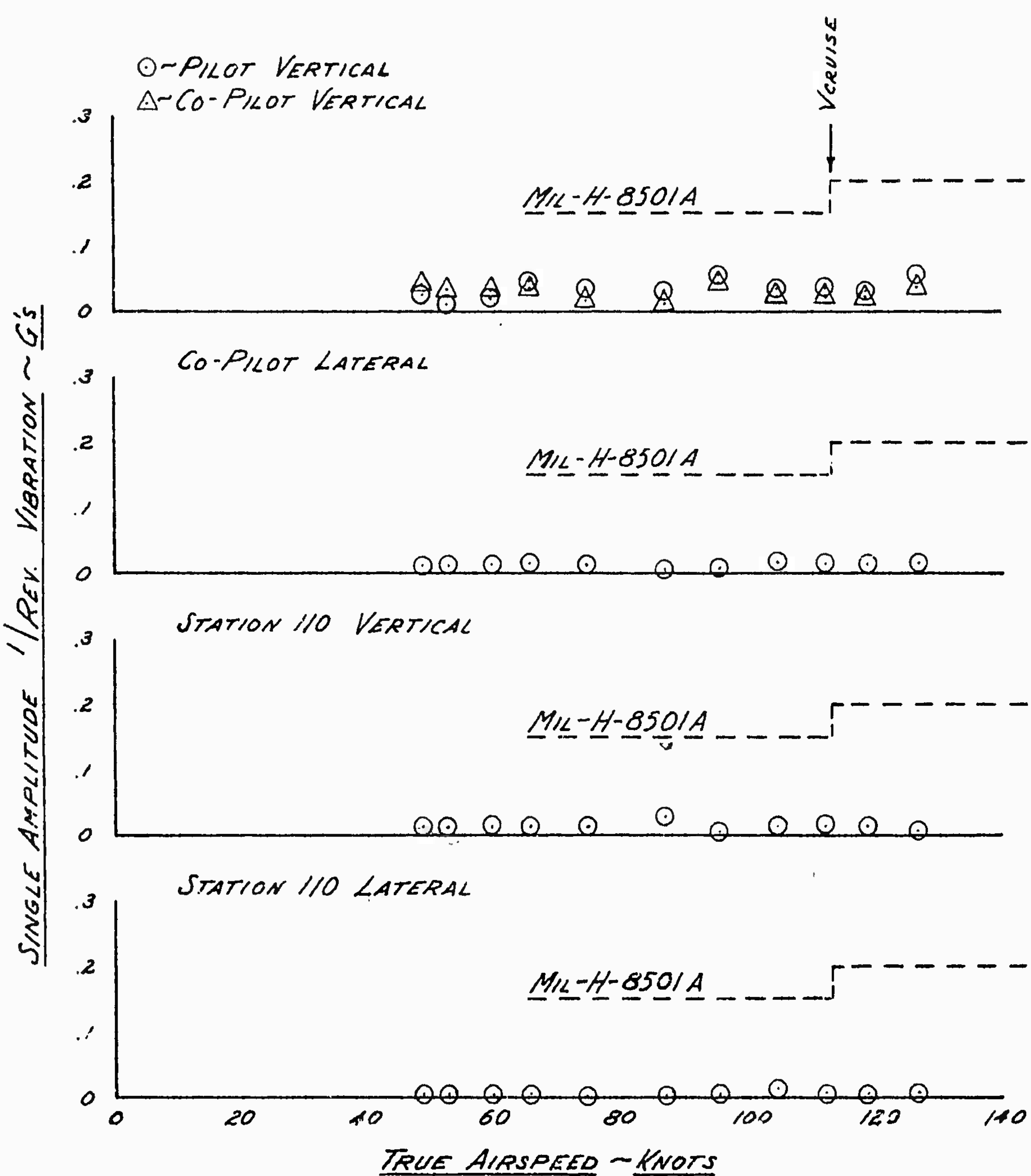


FIGURE No. 15

VIBRATION CHARACTERISTICS

UH-1B USA S/N 63-8684

BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 7480 LB
DENSITY ALTITUDE ~ 5490 FT.
ROTOR SPEED ~ 325.5 RPM
C.G. ~ STATION 126.0 (FWD)

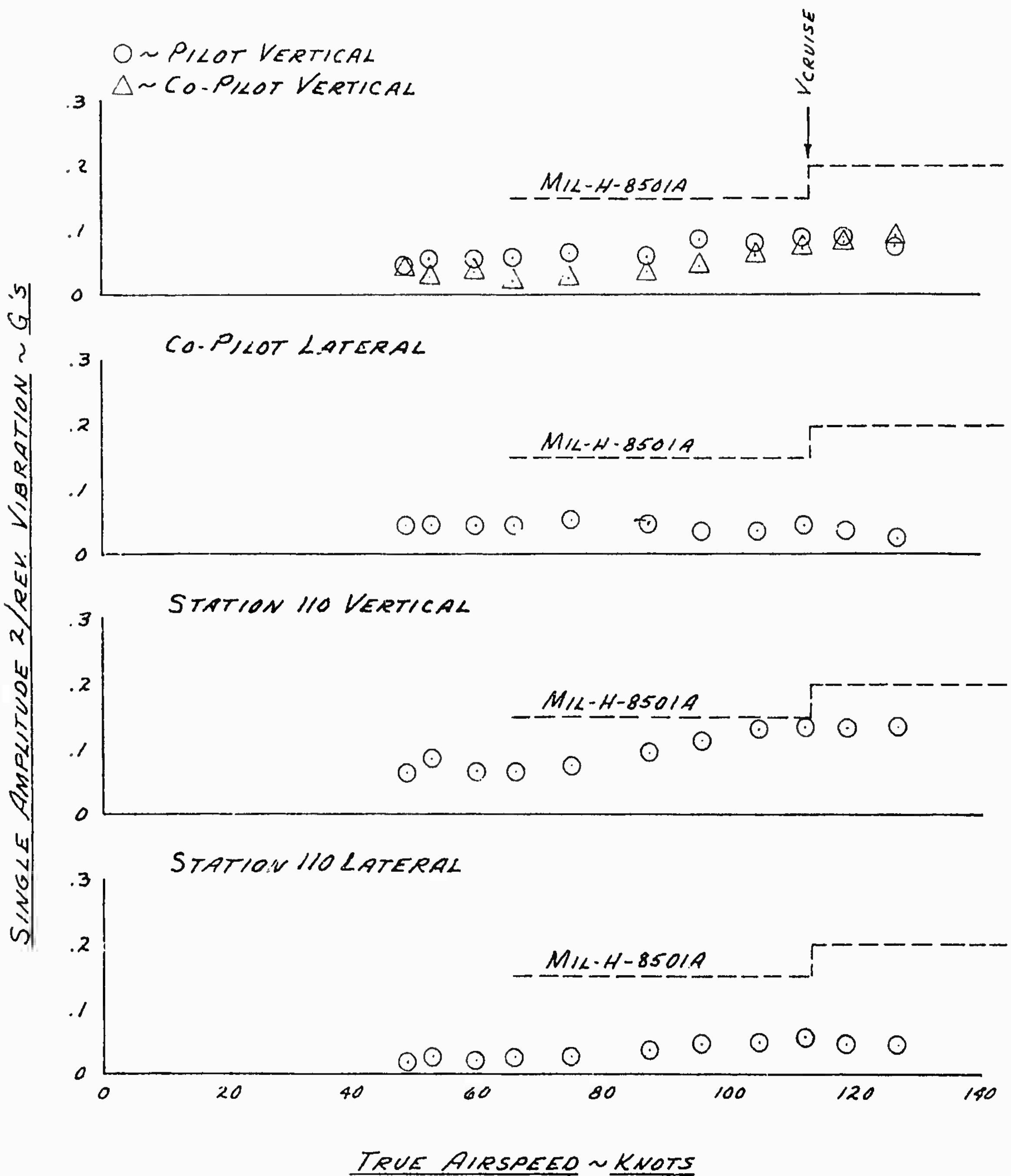


FIGURE NO. 16
VIBRATION CHARACTERISTICS
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 7480 LB
DENSITY ALTITUDE ~ 5490 FT.
ROTOR SPEED ~ 325.5 RPM
C.G ~ STATION 126.0 (FWD)

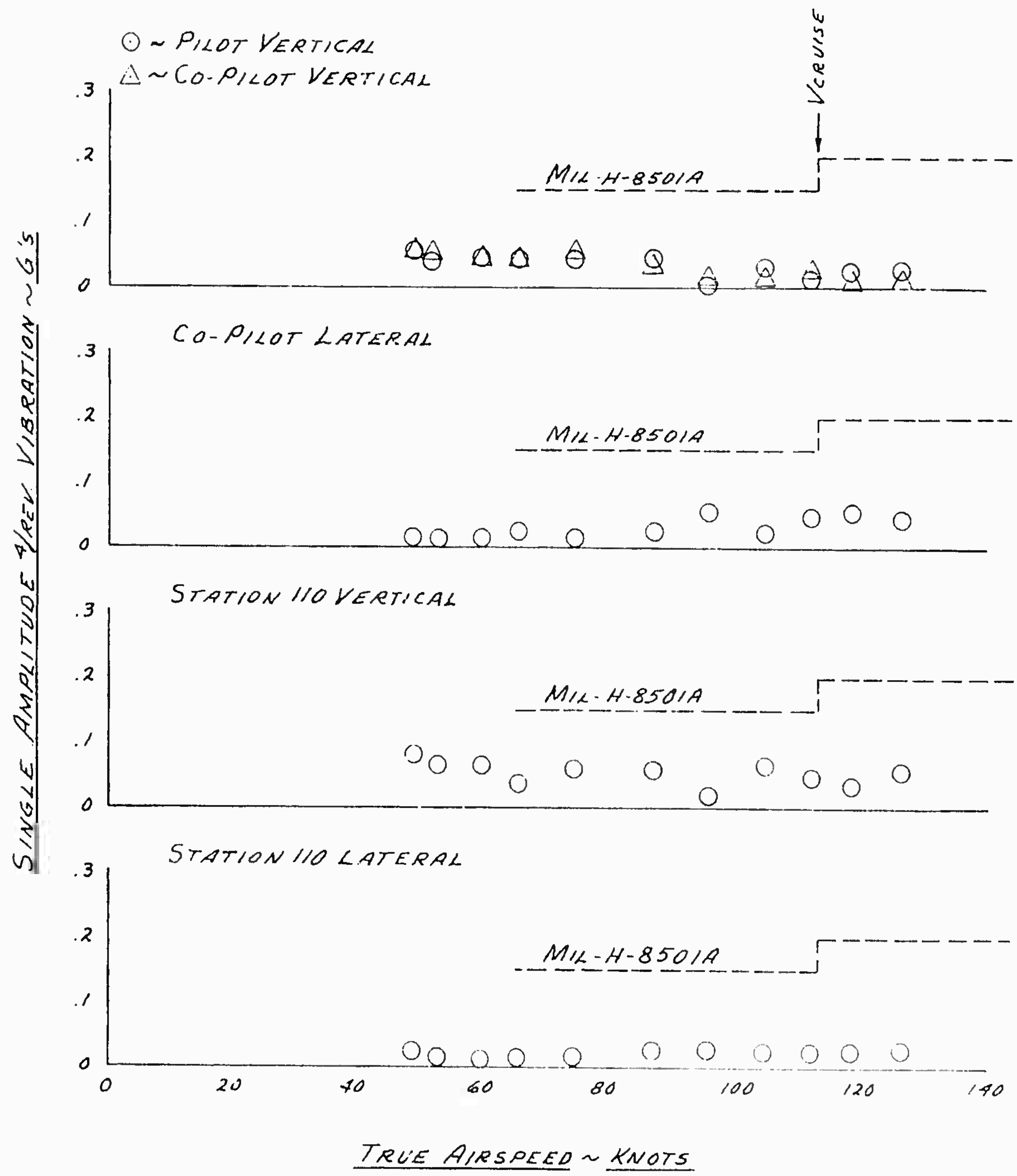


FIGURE NO. 17
VIBRATION CHARACTERISTICS
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 7270 LB
DENSITY ALTITUDE ~ 6240 FT.
ROTOR SPEED ~ 324.5 RPM
C.G. ~ STATION 131.0 (MID)

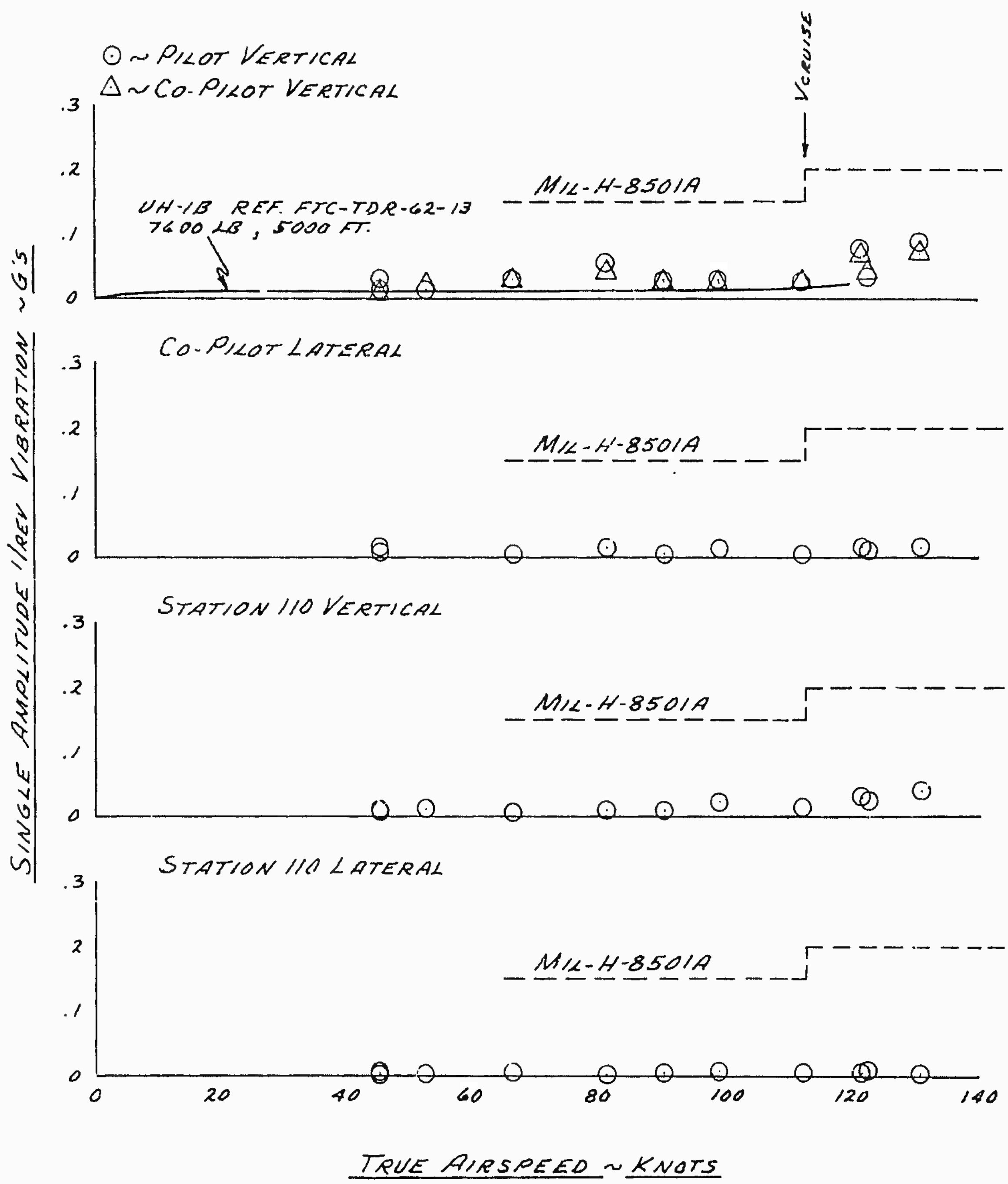


FIGURE No. 18
VIBRATION CHARACTERISTICS
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 7270 LB
DENSITY ALTITUDE ~ 6240 FT.
ROTOR SPEED ~ 324.5 RPM
C.G. ~ STATION 131.0 (MID)

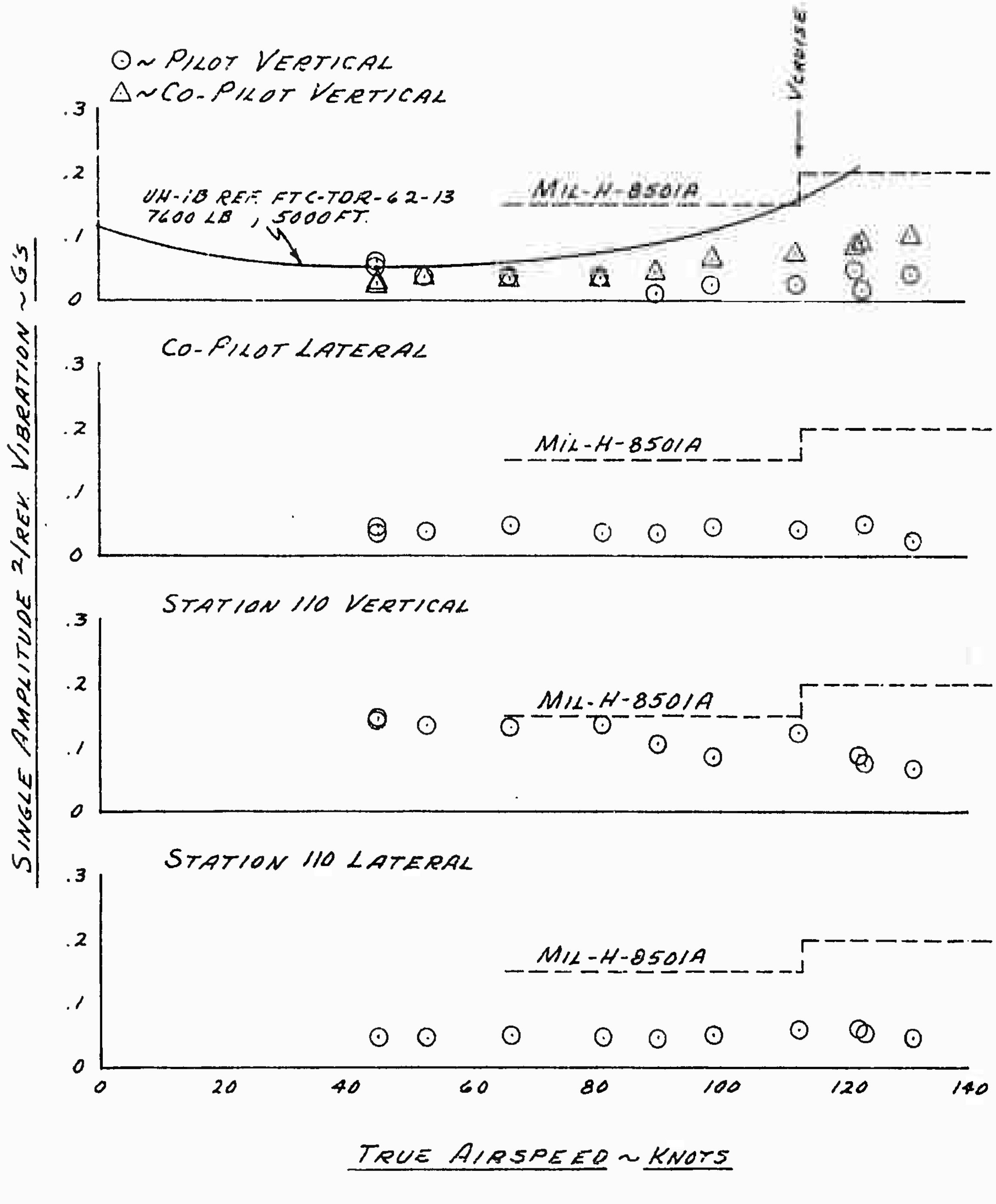


FIGURE NO. 19

VIBRATION CHARACTERISTICS

UH-1B USA S/N 63-8684

BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 7270 LB
DENSITY ALTITUDE ~ 6240 FT.
ROTOR SPEED ~ 324.5 RPM
C.G. ~ STATION 131.0 (MID)

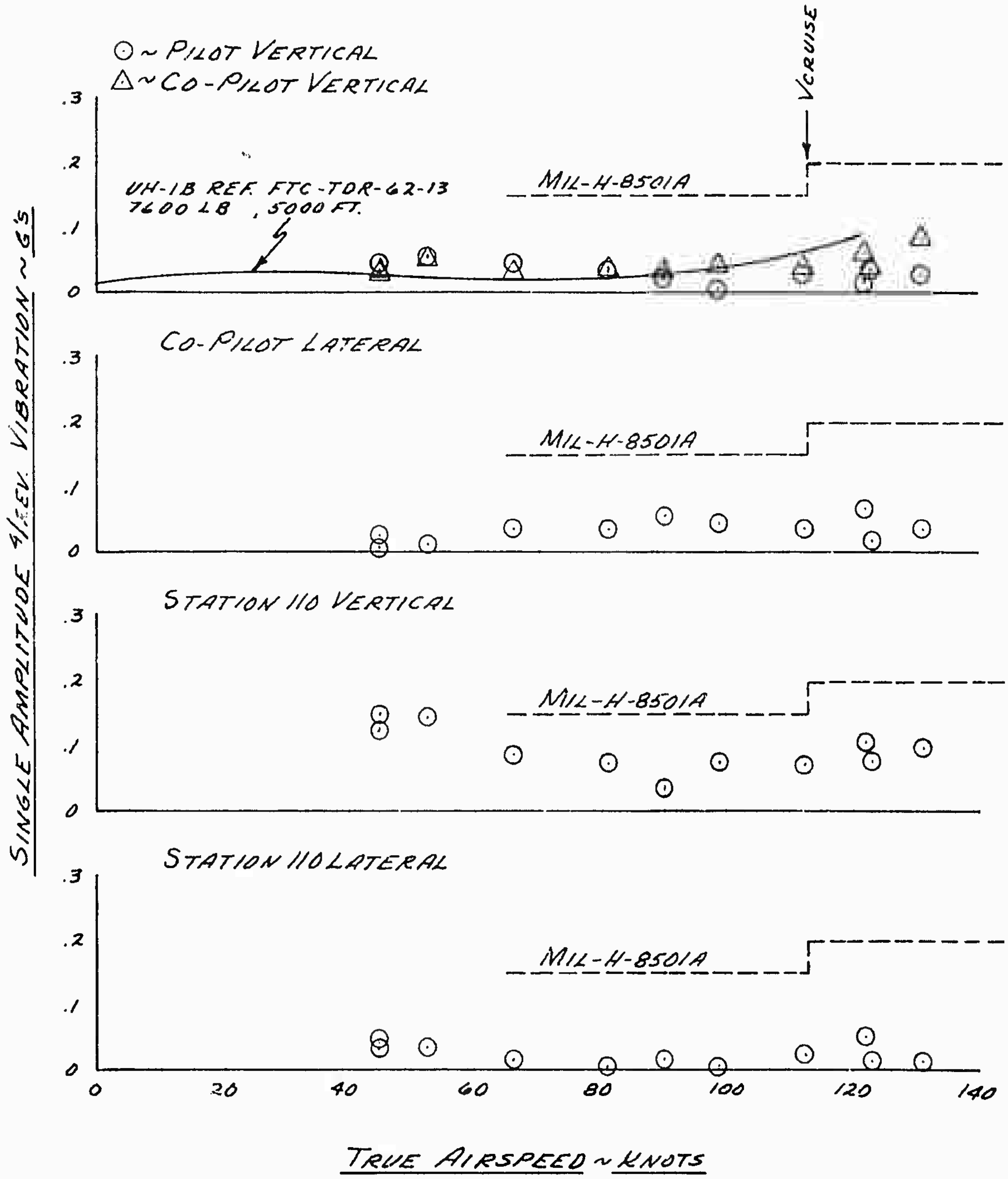


FIGURE NO. 20
VIBRATION CHARACTERISTICS
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 7490 LB
DENSITY ALTITUDE ~ 5140 FT.
ROTOR SPEED ~ 325.0 RPM
C.G. ~ STATION 136.0 (AFT)

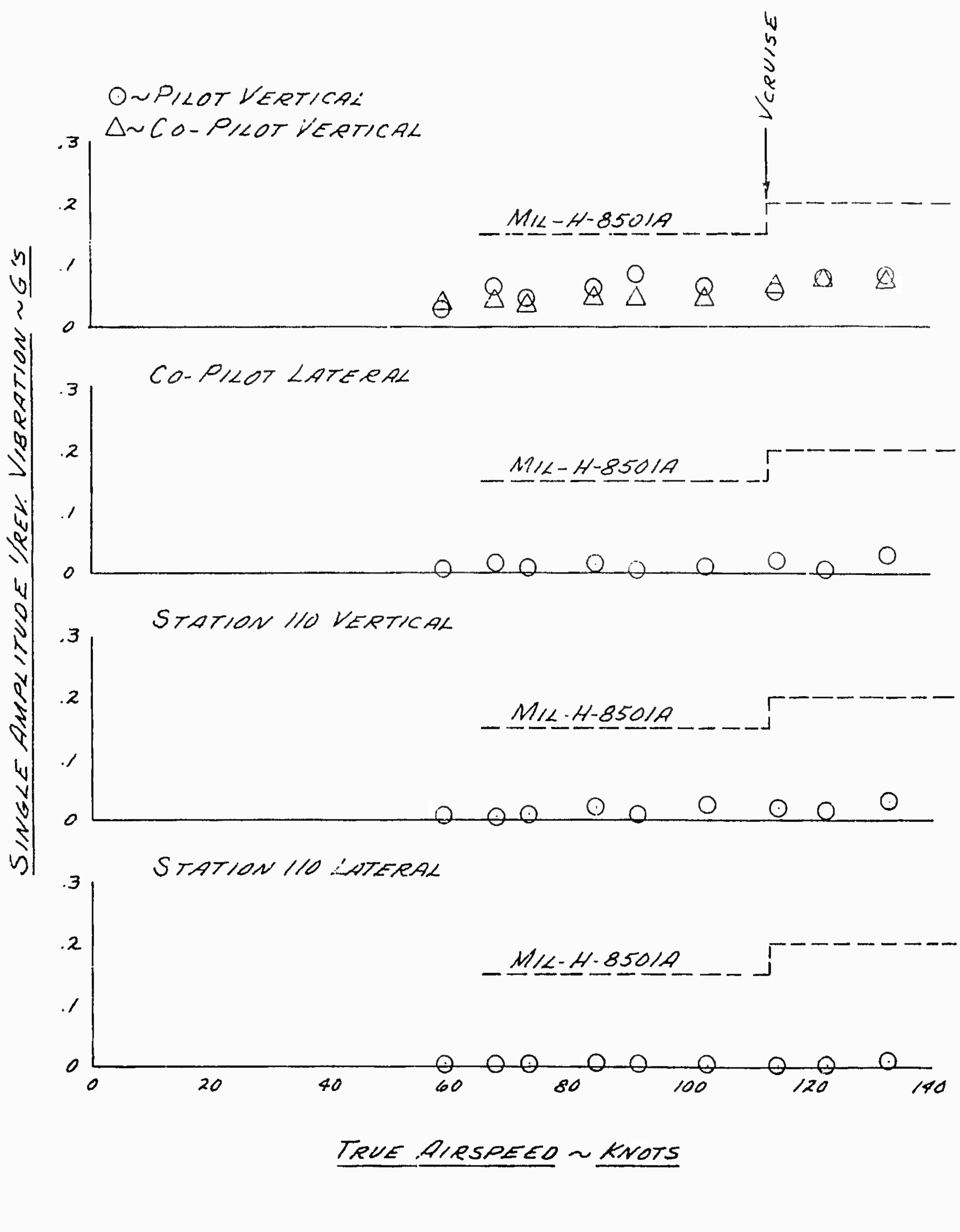


FIGURE NO. 21
VIBRATION CHARACTERISTICS
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 7490 LB
DENSITY ALTITUDE ~ 5140 FT.
ROTOR SPEED ~ 325.0 RPM
CG ~ STATION 136.0 (AFT)

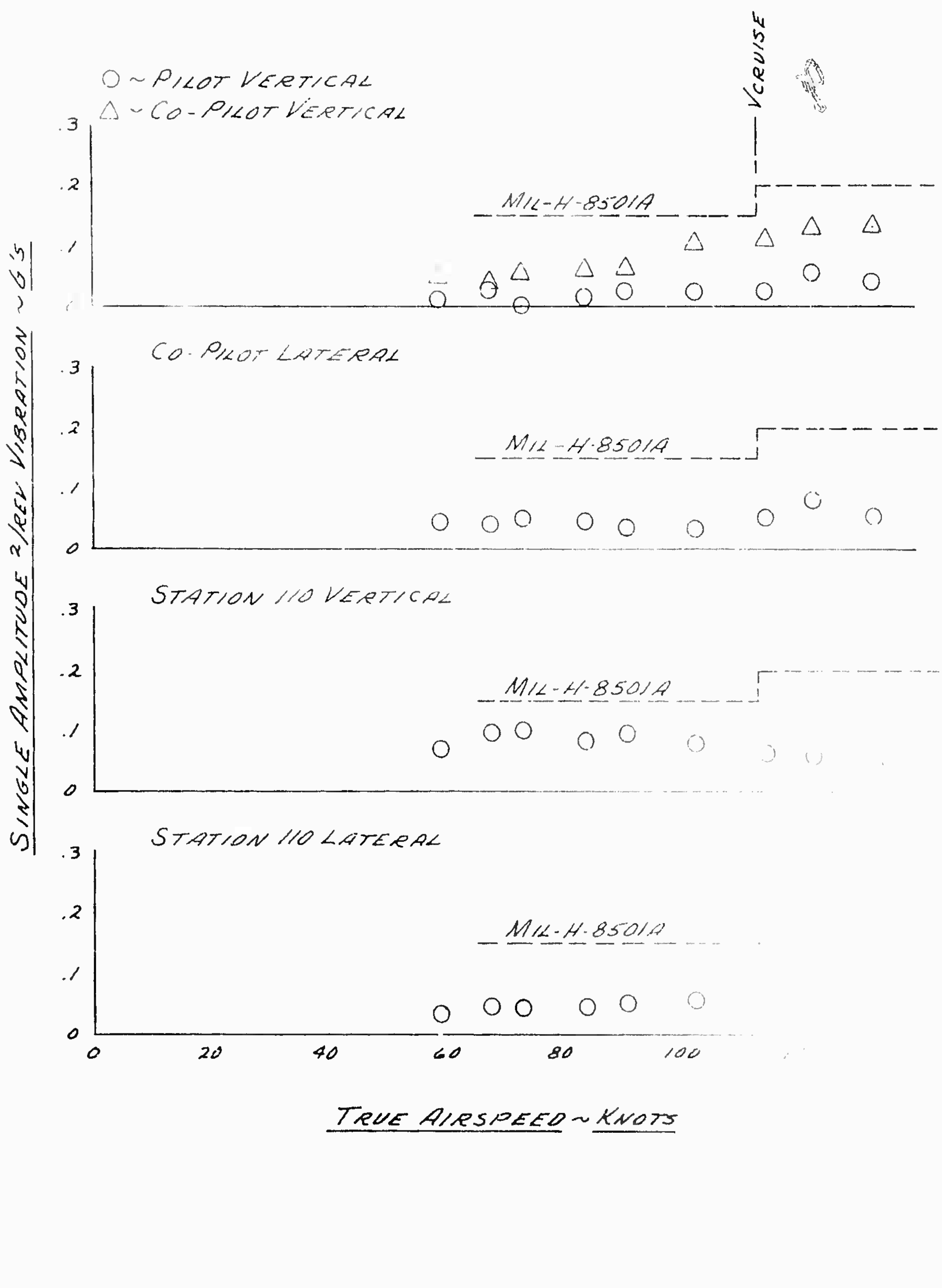


FIGURE NO. 22
VIBRATION CHARACTERISTICS
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 7490 LB
DENSITY ALTITUDE ~ 5140 FT.
ROTOR SPEED ~ 325.0 RPM
C.G. ~ STATION 136.0 (AFT)

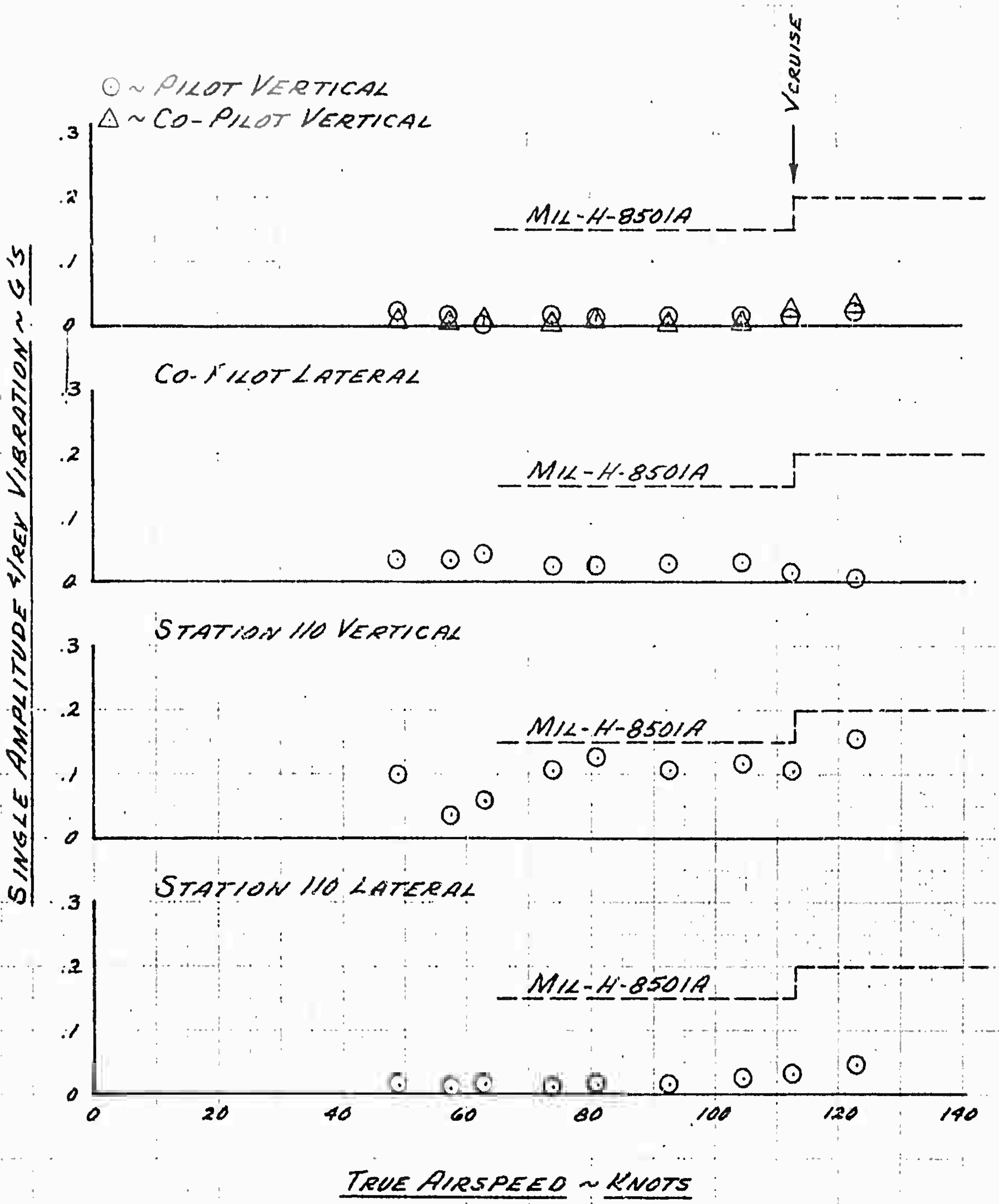


FIGURE No. 23

VIBRATION CHARACTERISTICS

UH-1B USA S/N 63-8684

BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 9160 LB
DENSITY ALTITUDE ~ 8980 FT.
ROTOR SPEED ~ 324.0 RPM
C.G. ~ STATION 131.0 (MID)

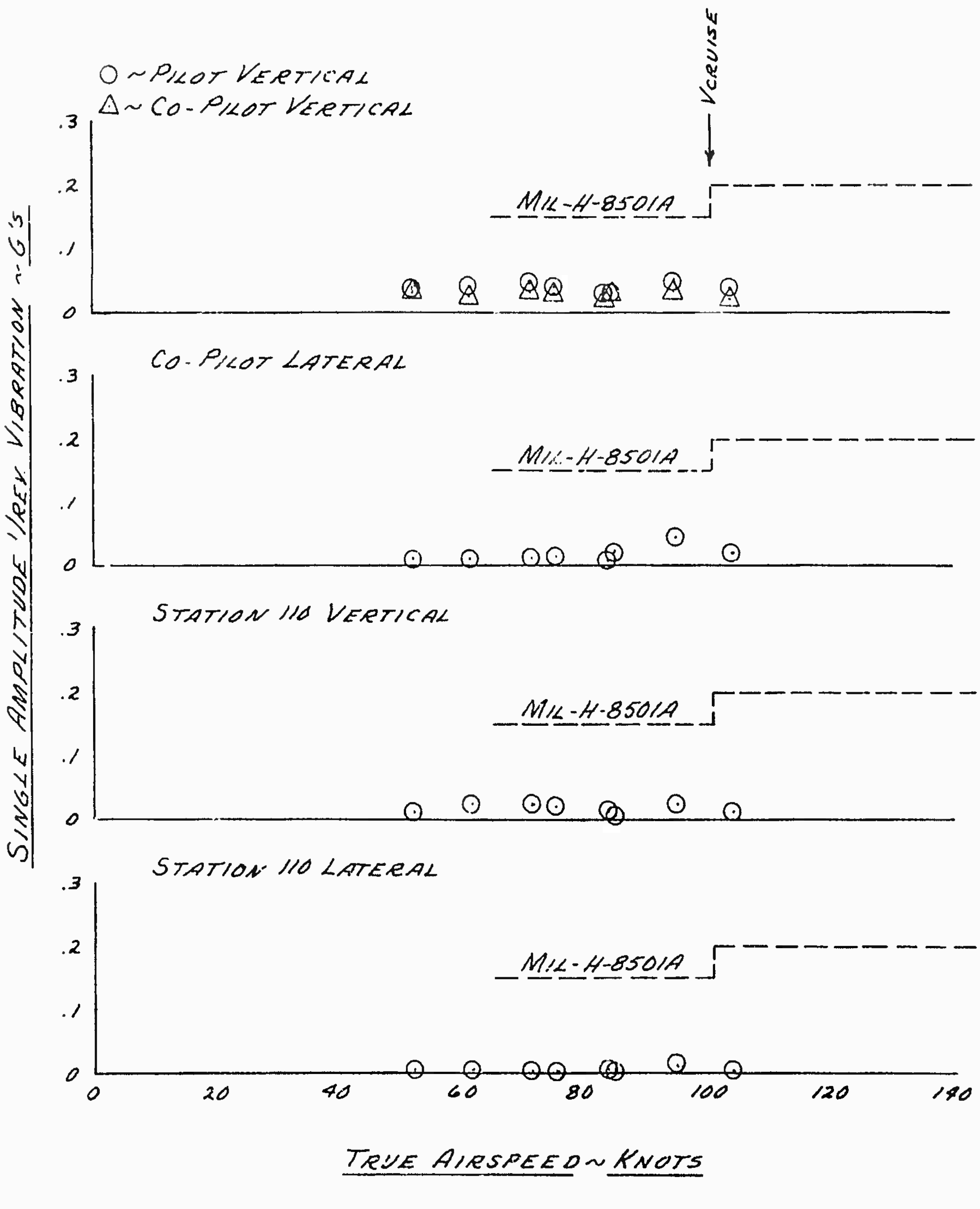


FIGURE NO. 24
VIBRATION CHARACTERISTICS
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 9160 LB
DENSITY ALTITUDE ~ 8980 FT.
ROTOR SPEED ~ 324.0 RPM
C.G. ~ STATION 131.0 (MID)

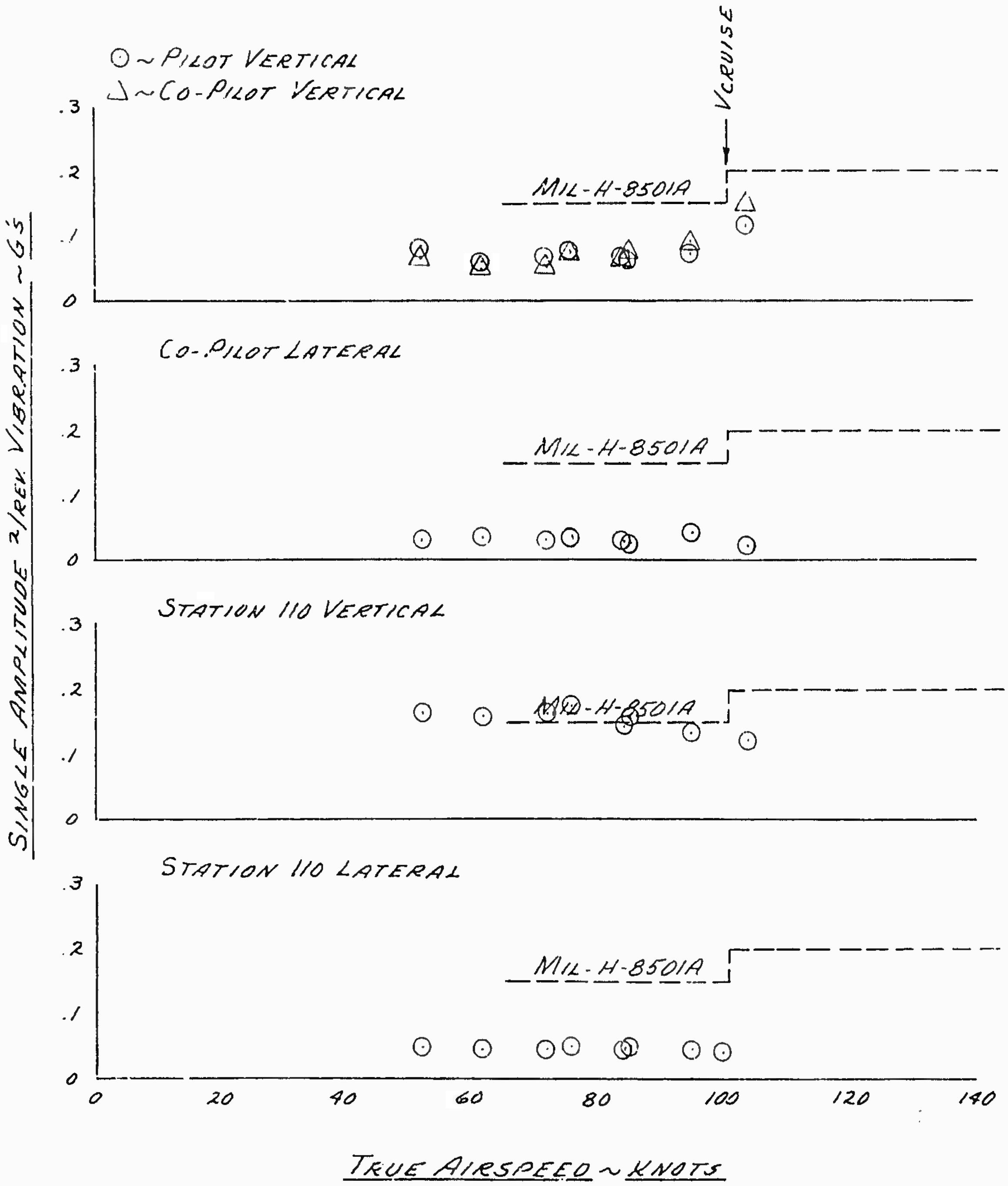


FIGURE NO. 25

VIBRATION CHARACTERISTICS

UH-1B USA S/N 63-8684

BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 9160 LB

DENSITY ALTITUDE ~ 8980 FT

ROTOR SPEED ~ 324.0 RPM

C.G. ~ STATION 131.0 (MID)

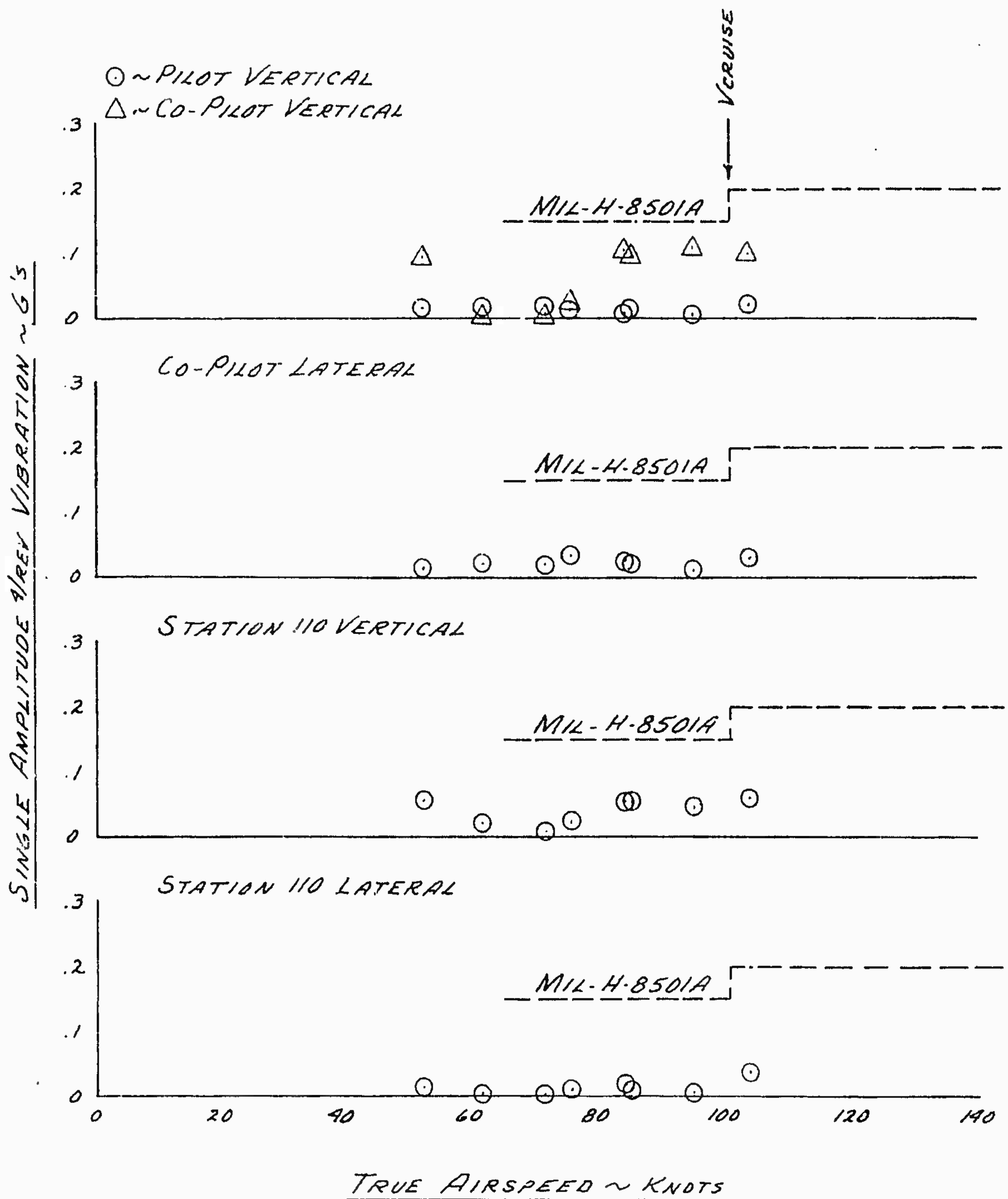


FIGURE NO. 26
AUTOROTATION PERFORMANCE
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 7400 LB
ROTOR SPEED ~ 323.0 RPM
C.G. ~ STATION 131.0 (MID)

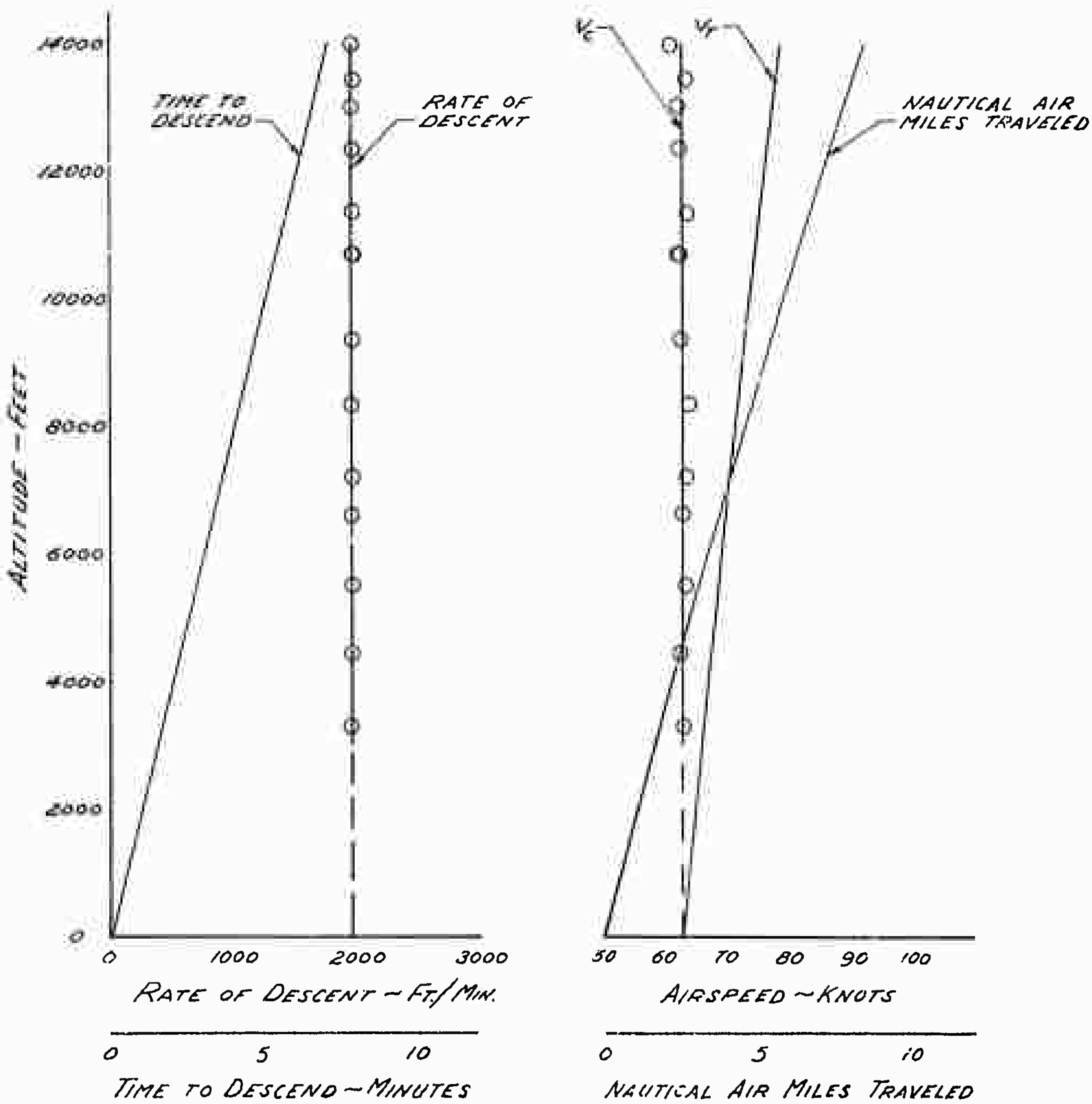


FIGURE NO. 27
CONTROL POSITIONS IN STABILIZED LEVEL FLT.
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

SYM.	G.W.~LB	H_D~FT.	RPM	C.G.
⊙	7760	5490	324	126 (FWD)
⊡	7660	6300	324	131 (MID)
△	7785	5135	325	136 (AFT)

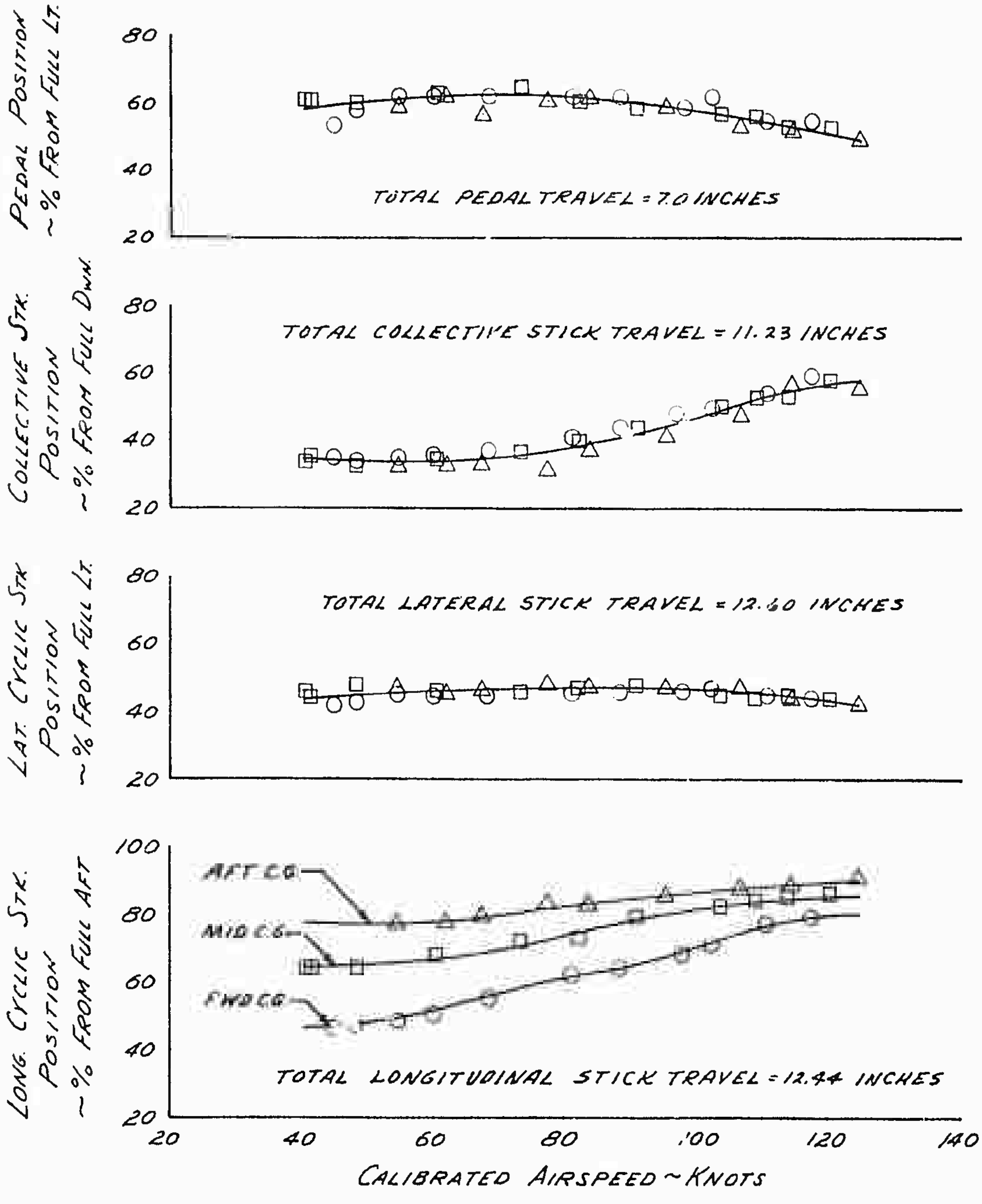


FIGURE NO. 28
STATIC LONGITUDINAL STABILITY
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 9300 LB
DENSITY ALTITUDE ~ 4000 FT.
ROTOR SPEED ~ 324.0 RPM
C.G. ~ STATION 126.6 (FWD)

NOTE:
SOLID SYMBOLS DENOTE LEVEL FLIGHT TRIM POINTS

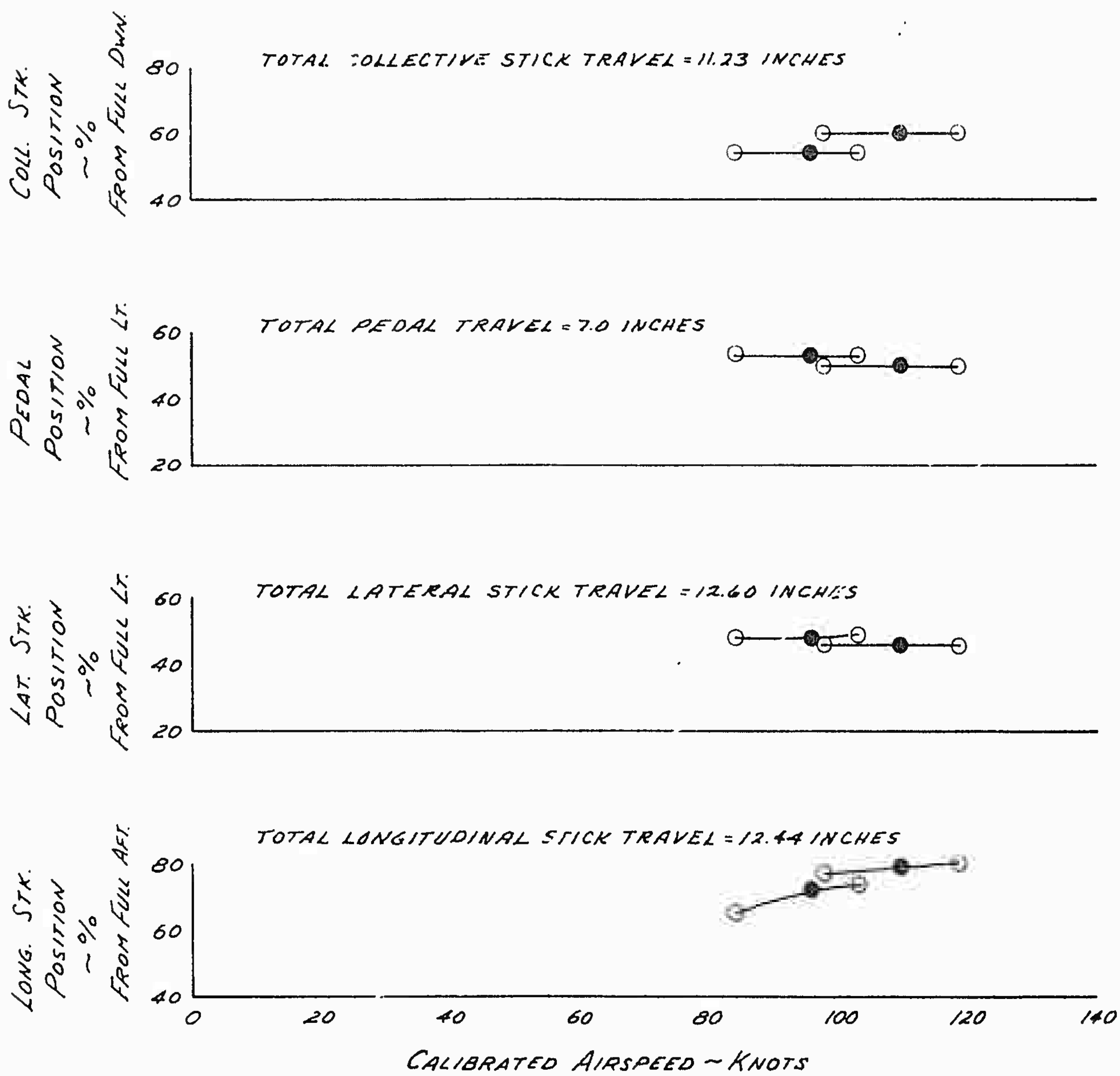


FIGURE No. 29
STATIC LONGITUDINAL STABILITY
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 6600 LB
DENSITY ALTITUDE ~ 5500 FT.
ROTOR SPEED ~ 324.0 RPM
C.G. ~ STATION 138.0 (AFT)

NOTE: SOLID SYMBOLS DENOTE LEVEL FLIGHT TRIM POINTS

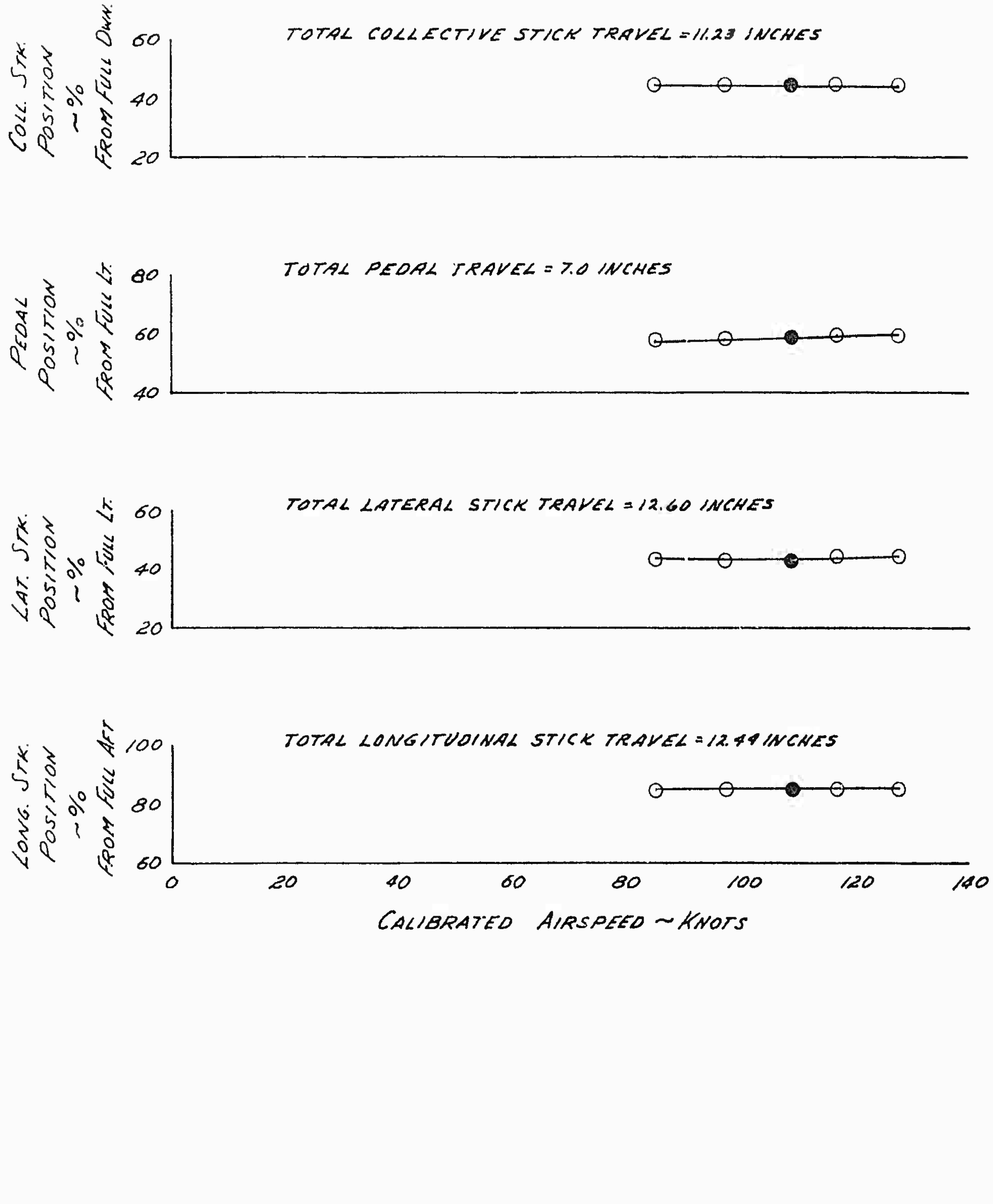


FIGURE NO. 30
TIME HISTORY OF CONTROL POSITIONS DURING THROTTLE CHOP
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 6600 LB
DENSITY ALTITUDE ~ 1300 FT.
C.G. ~ STATION 138.0 (AFT)
AIRSPEED ~ 131 KNOTS ~ V_c
THROTTLE CHOP WITH FLARE TO 60 KTS.

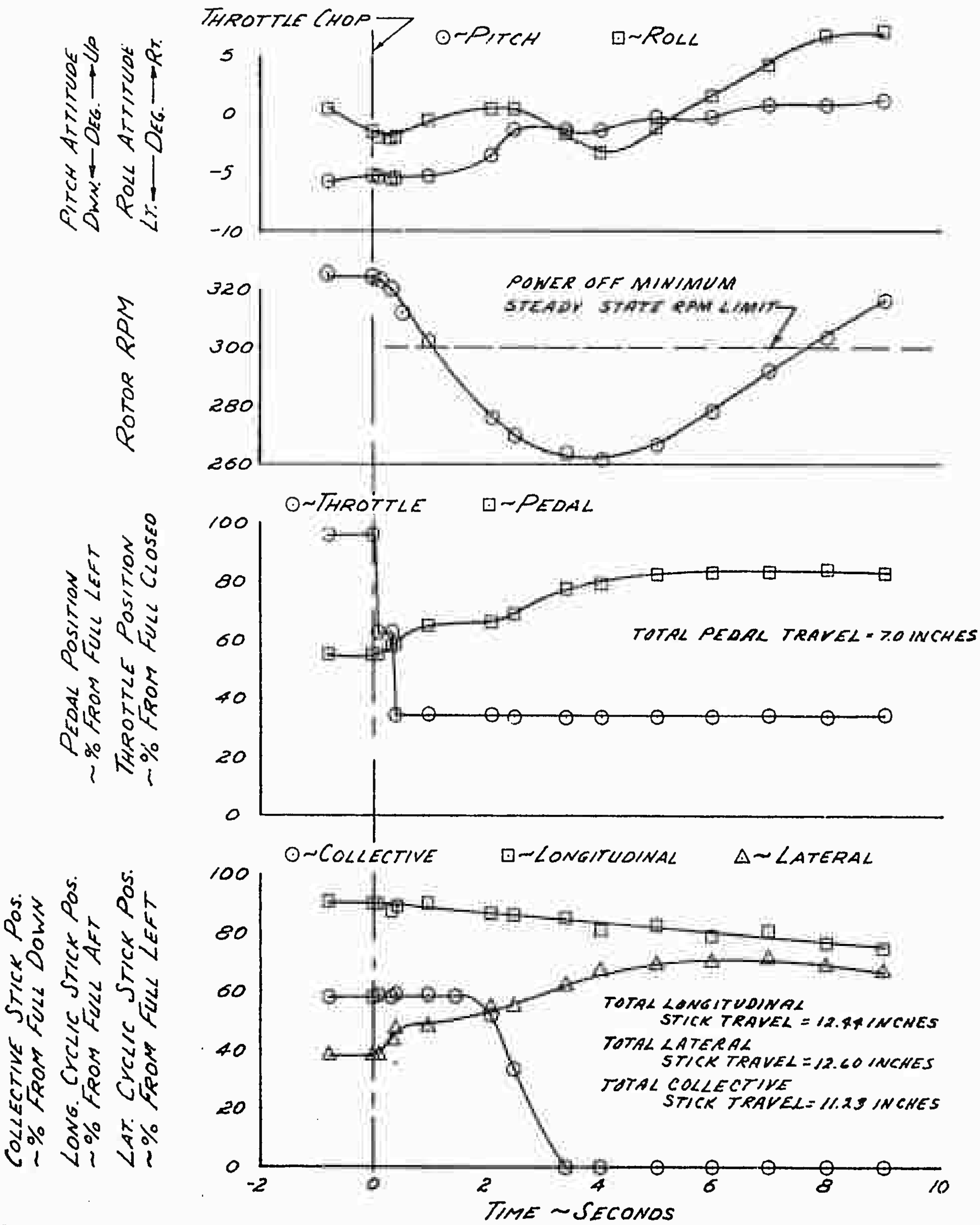


FIGURE NO. 31
TIME HISTORY OF CONTROL POSITIONS DURING THROTTLE CHOP
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 9020 LB
DENSITY ALTITUDE ~ 4070 FT
C.G. ~ STATION 126.6 (FWD)
AIRSPEED ~ 122 KNOTS ~ V_C
THROTTLE CHOP HOLDING CONSTANT ATTITUDE

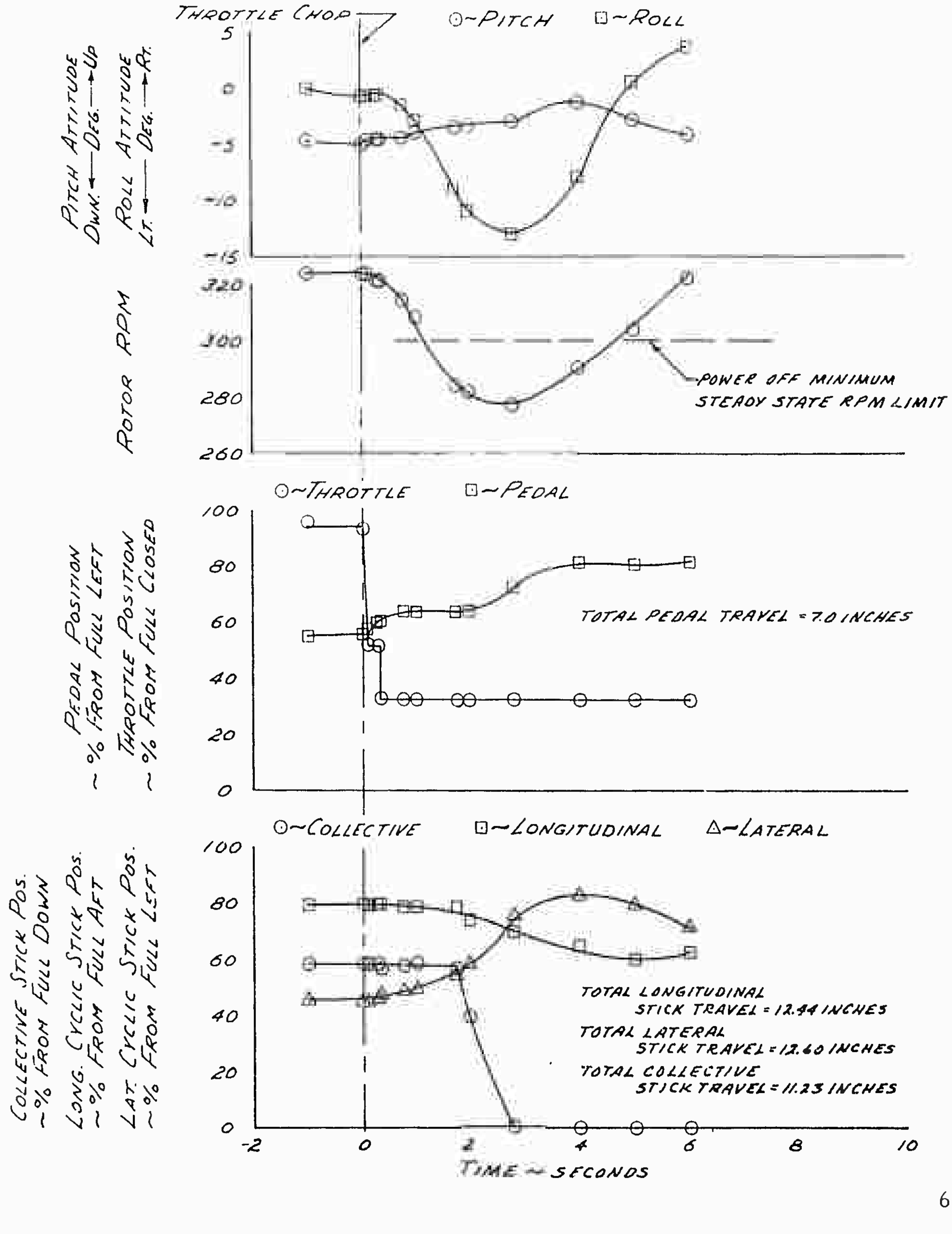


FIGURE NO. 32
TIME HISTORY OF CONTROL POSITIONS DURING THROTTLE CHOP
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

GROSS WEIGHT ~ 6600 LB
DENSITY ALTITUDE ~ 3650 FT.
C.G. ~ STATION 138.0 (AFT)
AIRSPEED ~ 131 KNOTS ~ V_C
THROTTLE CHOP HOLDING CONSTANT ATTITUDE

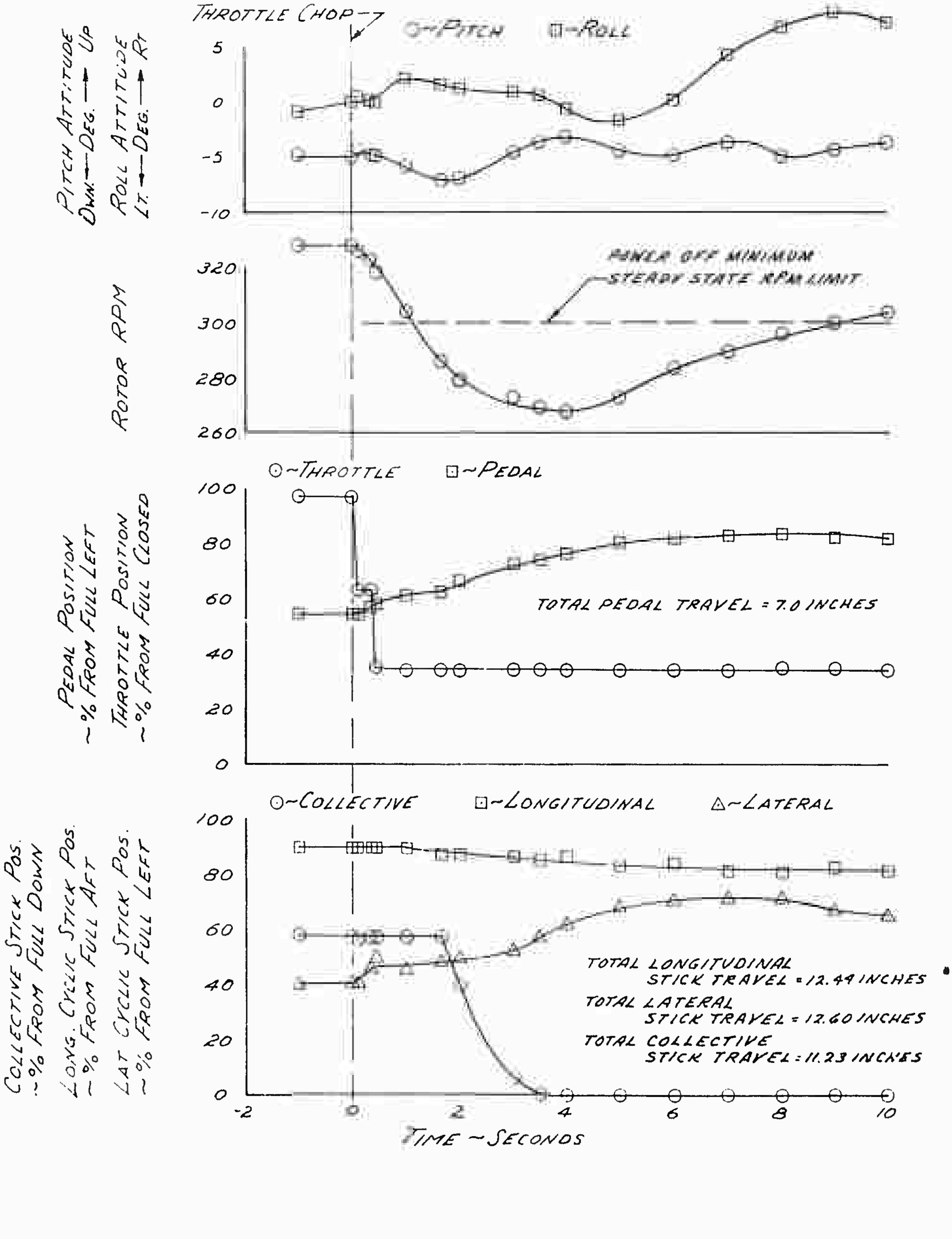


FIGURE NO. 33

AIRSPEED CALIBRATION

UH-1B USA S/N 63-8684

BELL MODEL 540 ROTOR SYSTEM

SHIPS SYSTEM

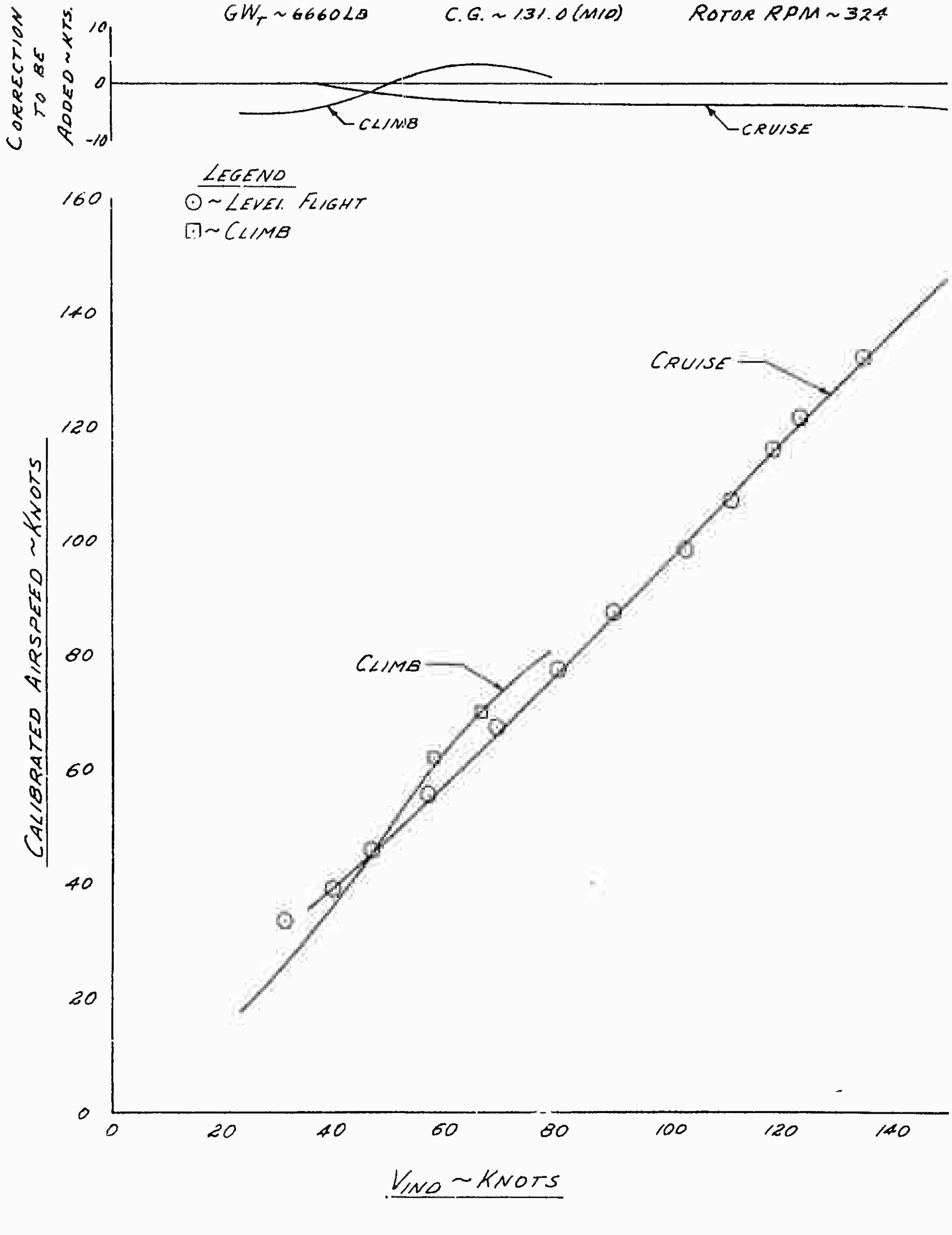


FIGURE NO. 34

INSTALLED POWER AVAILABLE

STANDARD DAY

T53-L-9 & 11 ENGINES

6600 RPM

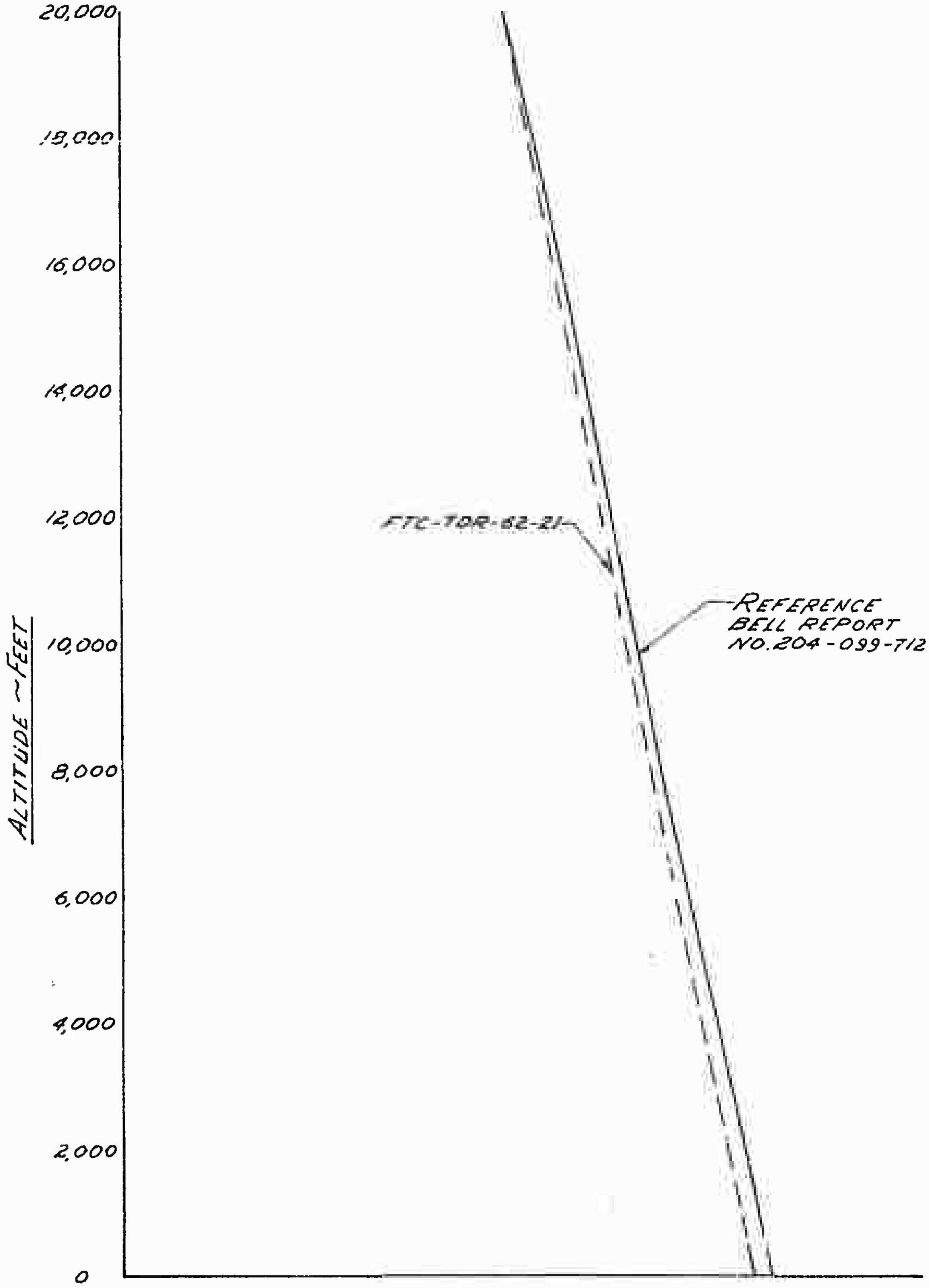
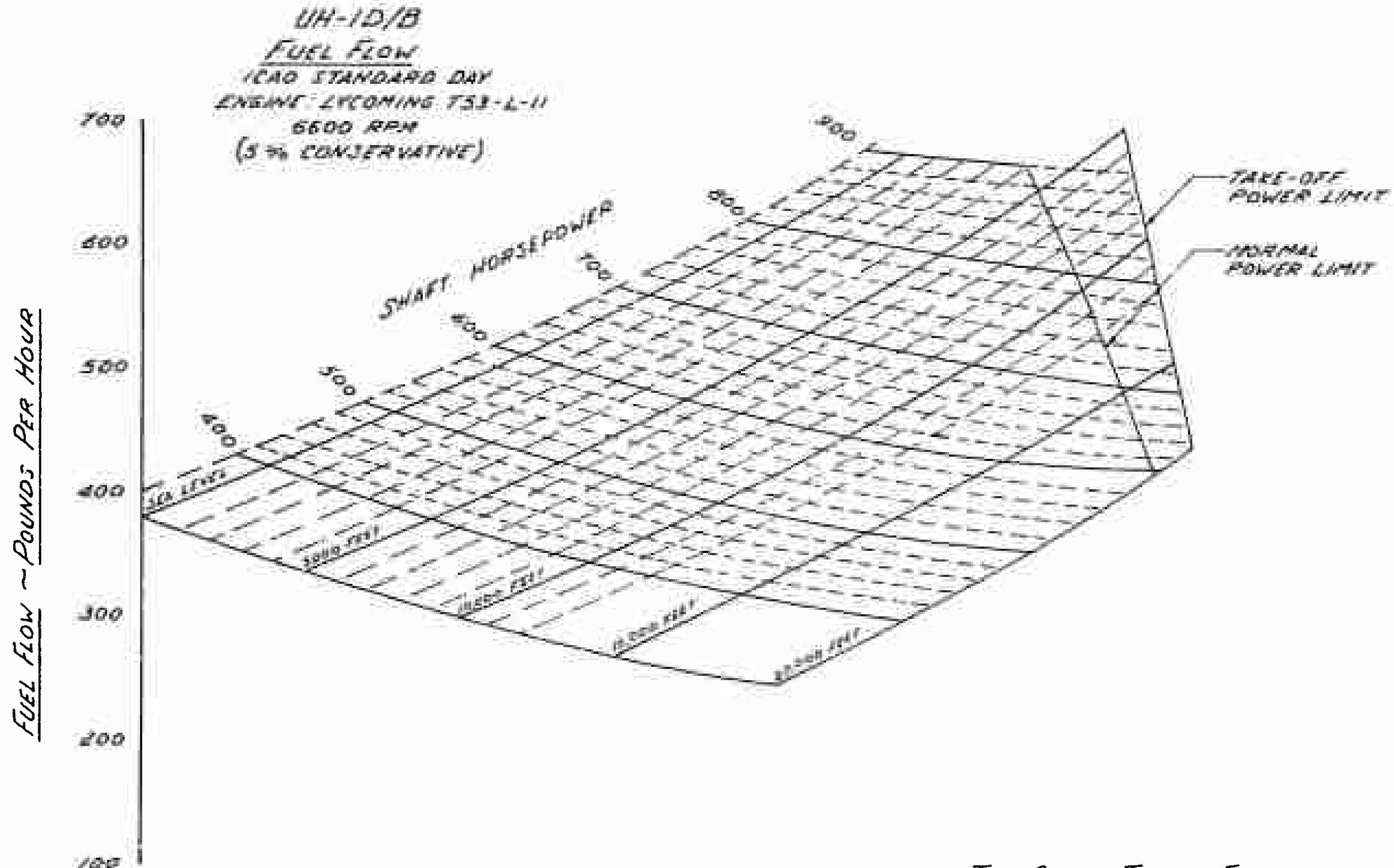


Figure No. 35
Specification Fuel Flow
UH-1B USA S/N 63-8684
Bell Model 540 Rotor System



This Curve Taken From
Bell Report 205-099-705

FIGURE No. 36

ENGINE TORQUE METER CALIBRATION

UH-1B USA S/N 63-8684

BELL MODEL 540 ROTOR SYSTEM

NOTE: DATA TAKEN FROM LYCOMING "GREEN RUN"
SHEETS DATED 2 JUNE 1964
ENGINE MODEL T-53-L-11
ENGINE SERIAL No. LEO 9542

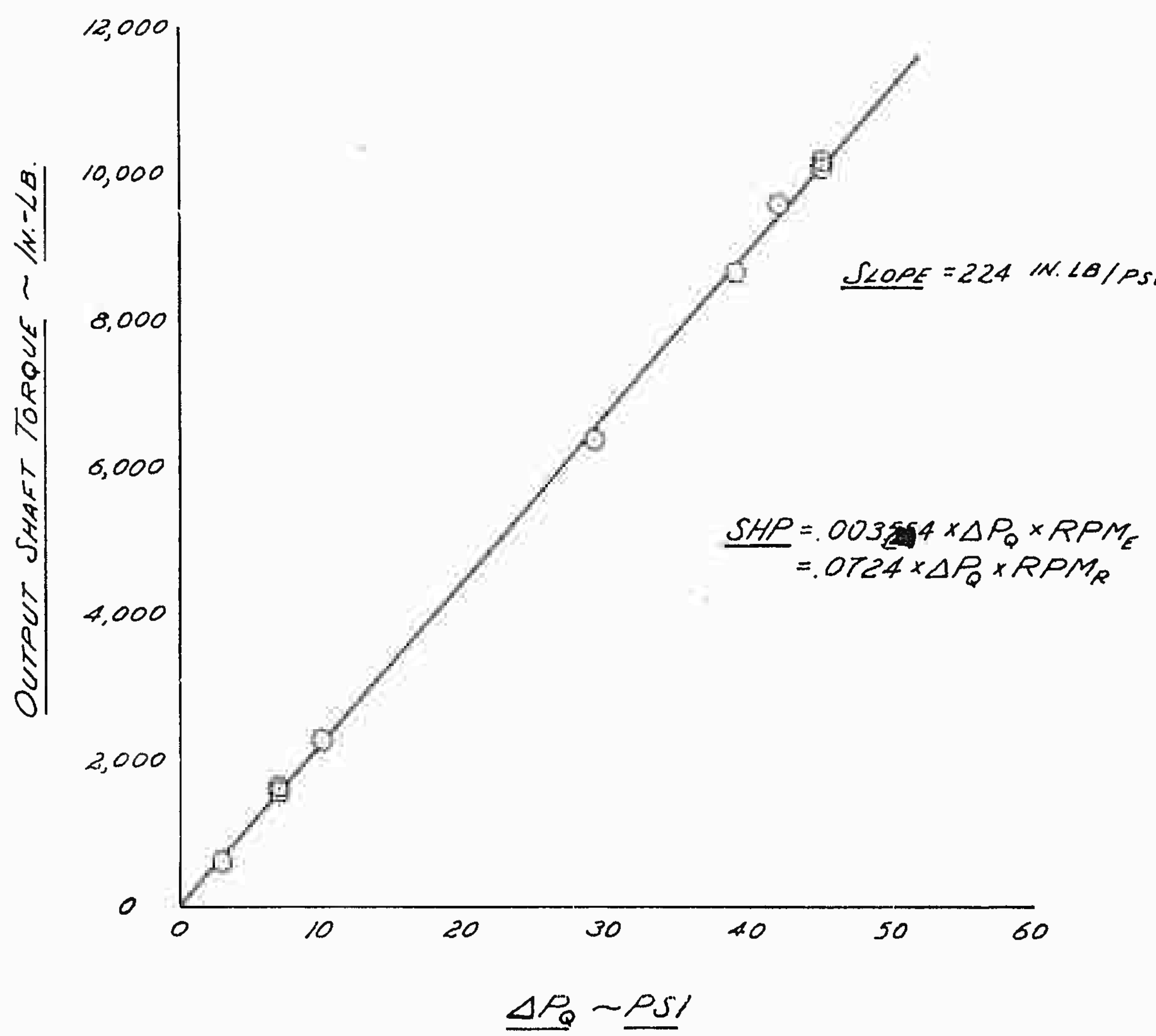
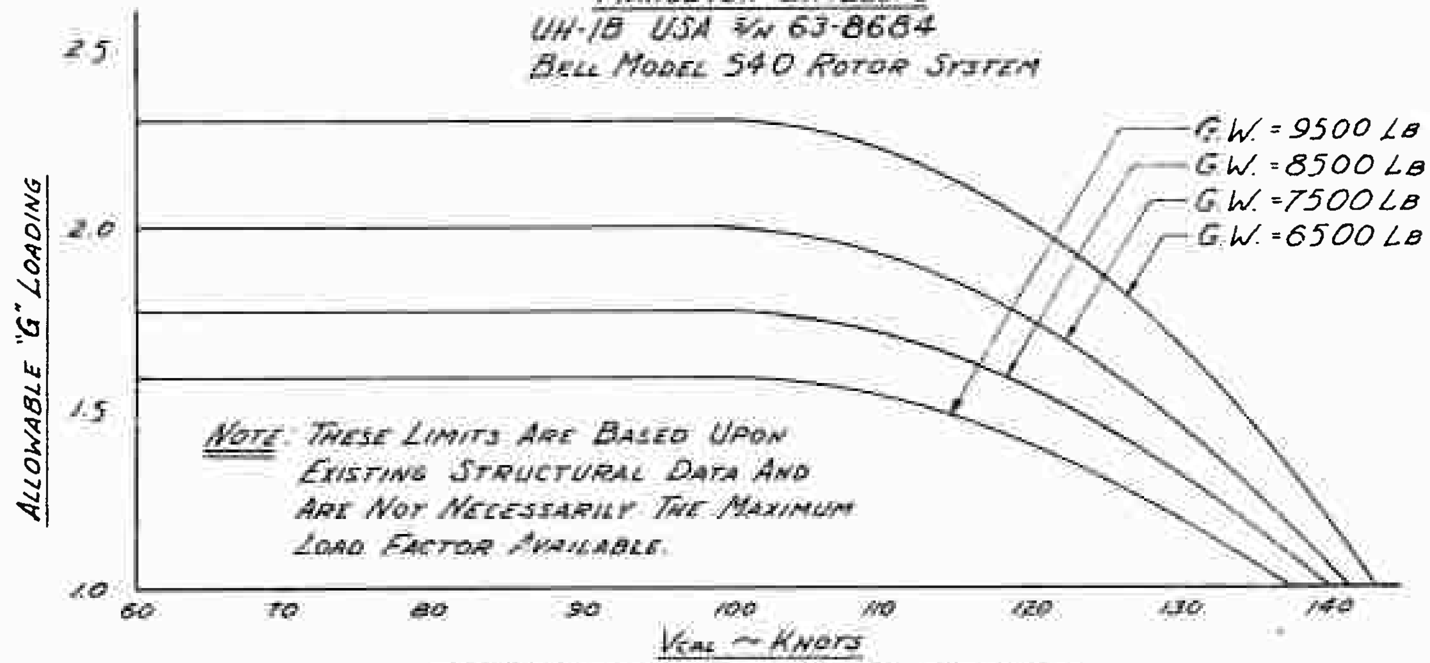
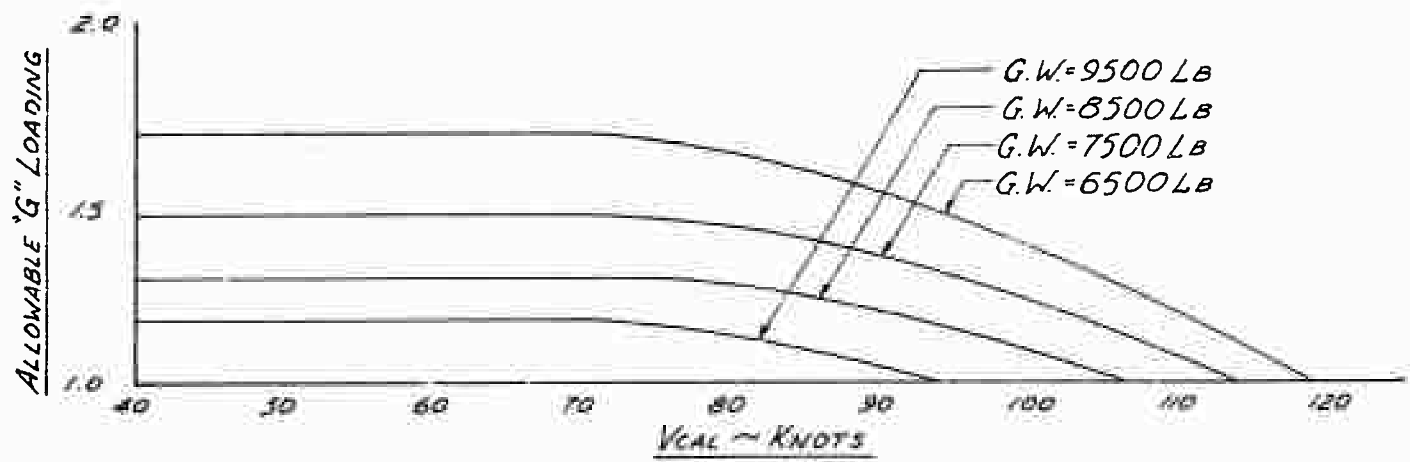


FIGURE NO. 37
MANUEVER ENVELOPE
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM



MANUEVER ENVELOPE AT S.L. ALTITUDE



MANUEVER ENVELOPE AT 10,000 FT. DENSITY ALTITUDE

Figure No. 38
C.G. Station vs Gross Weight
UH-1B USA S/N 63-8684
Bell Model 540 Rotor System

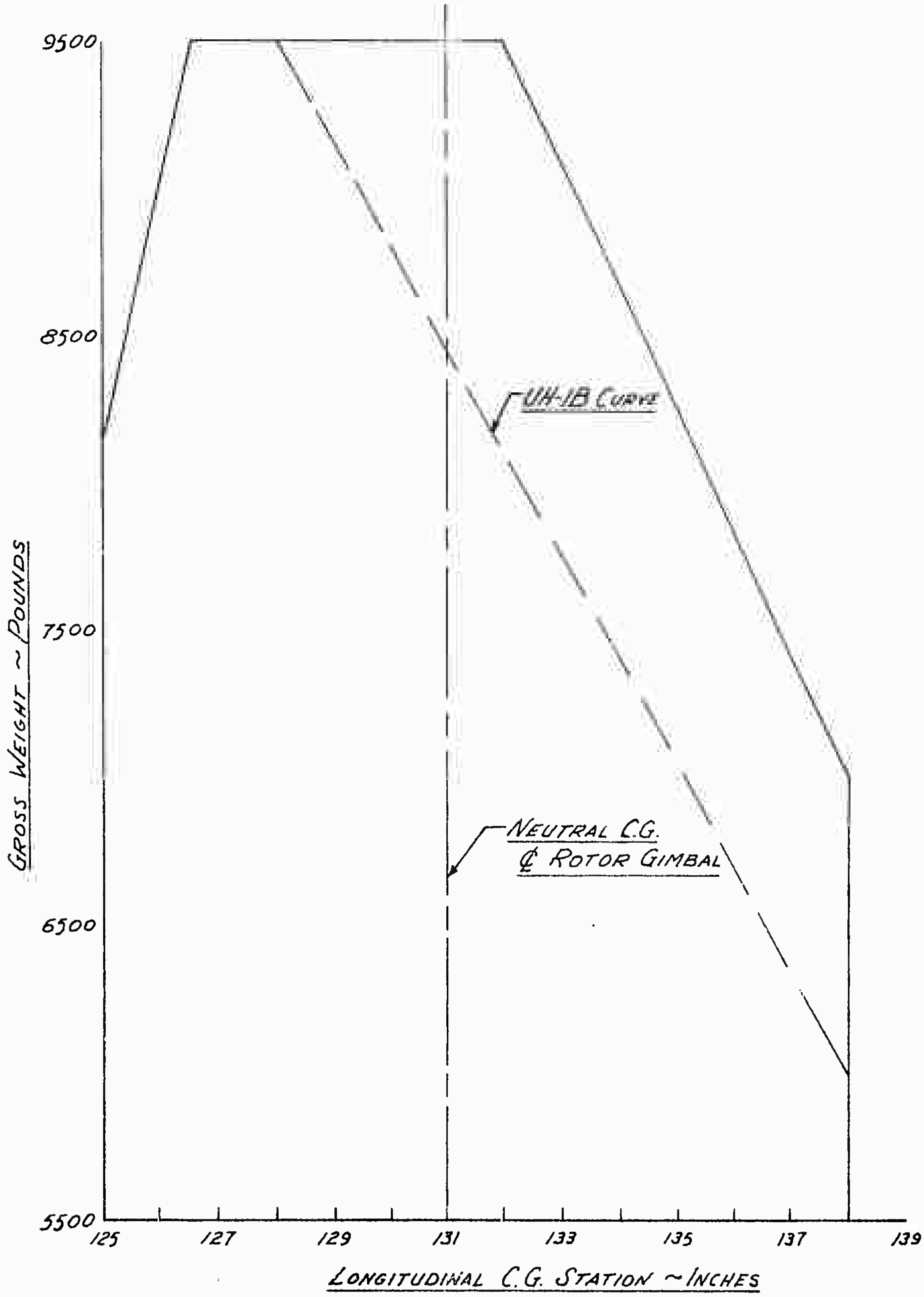


FIGURE NO. 39
SIDESLIP LIMITS
UH-1B USA S/N 63-8684
BELL MODEL 540 ROTOR SYSTEM

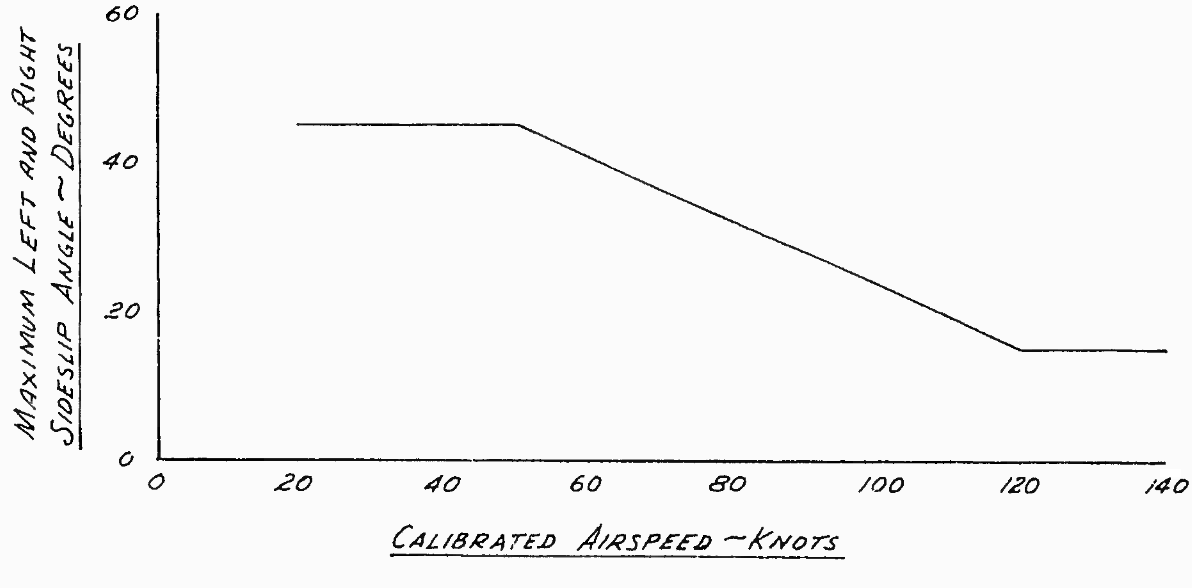
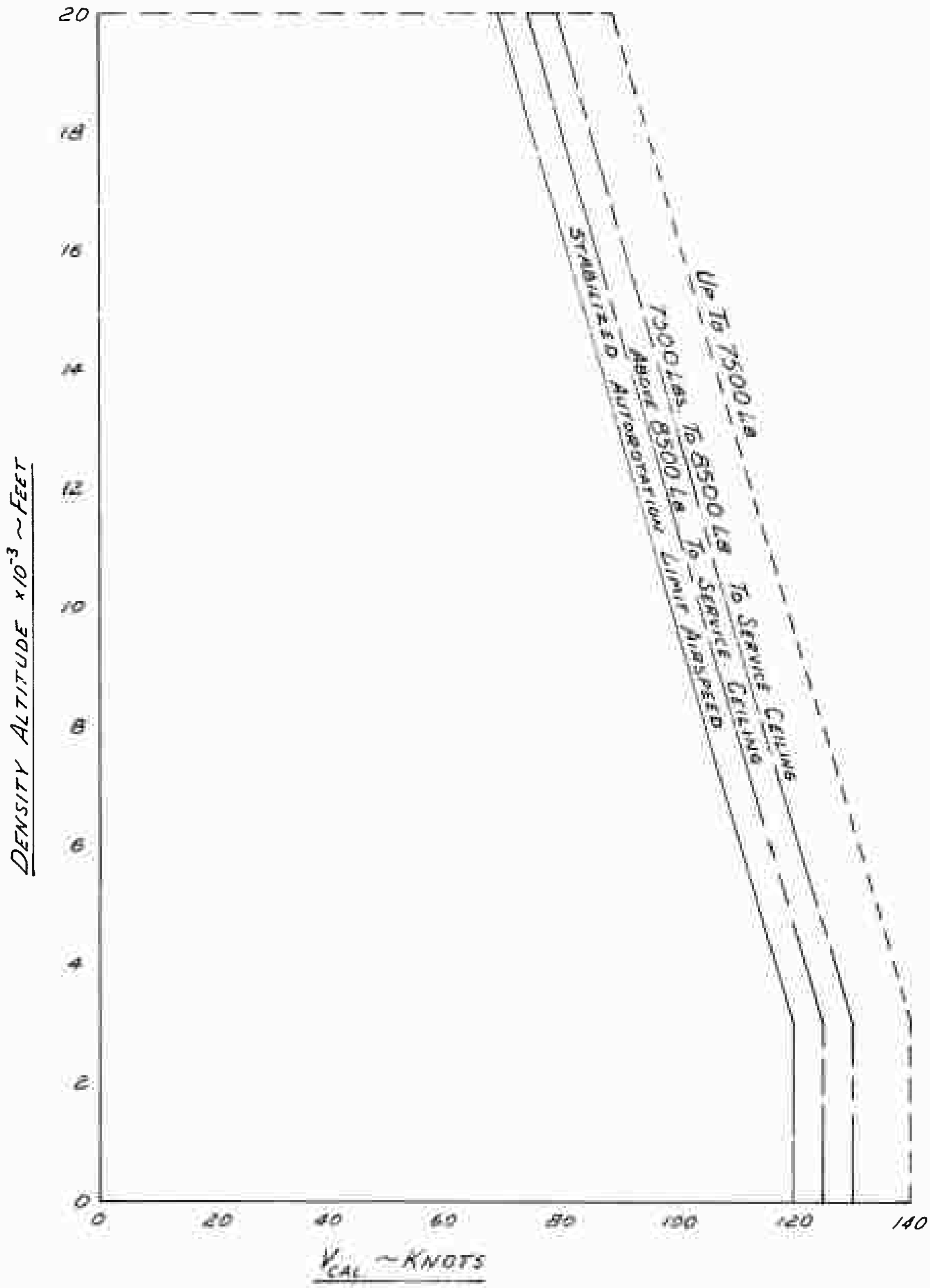


Figure No. 40
Vne Limits vs Density Altitude
UH-1B USA S/N 63-8684
Bell Model 540 Rotor System



APPENDIX II

DATA CALCULATION AND ANALYSIS METHODS

1.0 GENERAL

The test techniques employed and the data analysis methods required to correct the performance data from test conditions are described in this appendix.

Data analysis is generally based on use of the helicopter dimensionless performance parameters power coefficient (C_P), thrust coefficient (C_T), and advance ratio (μ). These parameters are defined by the following equations:

$$C_P = \frac{550\ SHP}{\rho A(\Omega R)^3} \quad \text{Coefficient of Power}$$

$$C_T = \frac{W}{\rho A(\Omega R)^2} \quad \text{Coefficient of Weight}$$

$$\mu = \frac{V_T \times 1.688}{\Omega R} \quad \text{Tip Speed Ratio}$$

The symbols used in this report are listed in the following table:

SYMBOL	DEFINITION	UNITS
SHP	Shaft Horsepower	550 ft-lb/sec
ρ	Atmospheric Density	Slugs/ft^3
ρo	Standard-Day Sea-Level Atmospheric Density	Slugs/ft^3
σ	Atmospheric Density Ratio	
A	Rotor Disc Area	ft^2
Ω	Rotor Angular Velocity	radians/sec
R	Rotor Radius	ft

SYMBOL	DEFINITION	UNITS
W	Gross Weight	lb
V	Airspeed	kt
R/C	Rate of Climb (tapeline)	ft/min
dH_p/dt	Slope of Pressure Altitude vs Time Plot	ft/min
R/D	Rate of Descent (tapeline)	ft/min
T	Temperature	deg K
N_I	Gas Producer Speed	rpm
N_{II}	Power Turbine Speed	rpm
θ	Temperature Ratio	
δ	Pressure Ratio	
W_f	Fuel Flow	lb/hr
G, N	Load Factor, Acceleration	
t	Time	hr, min, sec
C_P	Power Coefficient	
C_T	Thrust Coefficient	
K_P	Power Constant	
K_W	Weight Constant	

SUBSCRIPT	DEFINITION
Δ	Increment to be Added
t	Test Condition
s	Standard Condition
a, o	Ambient Condition

SUBSCRIPT	DEFINITION
T	True Airspeed
C	Calibrated Airspeed
q	Torque Pressure
w	Weight

2.0 POWER DETERMINATION

2.1 POWER REQUIRED

Power required data was obtained by means of calibrated rotor rpm and torquemeter instrumentation and the following equation:

$$SHP_t = RPM \times \Delta psi_q \times .0724$$

This constant .0724 was obtained from the slope of the torquemeter calibration curve (in-lb/psi) and the gear reduction between output shaft and the rotor.

2.2 POWER AVAILABLE

Standard-day power-available information was taken from Reference n. The installed power available from this report is presently the basis for the performance data in the UH-1B Operator's Manual (Reference p). The fuel flow information presented in Figure 35, Appendix I is similarly the present basis for range performance for the Operator's Manual.

The standard-day power available data presented in FTC-TDR-62-21 (Reference m) was also used to determine performance increments due to installations of the Model 540 rotor system compared with a standard UH-1B. For all comparisons, the fuel flow data of Figure 35 was used.

3.0 CLIMB PERFORMANCE

The observed rate of climb was corrected to tapeline by the expression:

$$R/C_t = \frac{dH_p}{dt} \times \frac{T_t}{T_s}$$

Power corrections were made by the use of:

$$\Delta R/C_p = K_p \times \frac{\Delta SHP\ 33,000}{W_t}$$

Where K_p = .670 (Reference m, FTC-TDR-62-21)

Weight corrections were made by the use of:

$$\Delta R/C_w = K_w\ SHP_s\ 33,000 \left(\frac{1}{W_s} - \frac{1}{W_t} \right)$$

Where K_w = .745 (Reference m, FTC-TDR-62-21)

4.0 LEVEL FLIGHT

During the level flight tests, density altitude was increased as fuel was consumed to maintain a near constant value of C_T.

The analyzed data was plotted as a function of true airspeeds; this is presented in Figures 7 through 13, Appendix I. Specific range was determined for each level flight curve by determining fuel flow from Figure 35 and making the appropriate calculations.

The data was cross-plotted in dimensionless C_P, C_T and μ form; this is presented in Figures 3 through 6.

The level flight summary, Figure 2, was obtained by selecting the recommended cruise condition values of Figures 7 through 13, calculating Range Factor (NAMPP x weight) and determining cruise airspeeds. The data was then plotted as a function of thrust coefficient, C_T.

5.0 AUTOROTATION

The observed rate of descent was corrected to tapeline by the expression:

$$R/D = \frac{dH_p}{dt} \times \frac{T_t}{T_s}$$

Rotor decay rates were determined from time history plots of rotor rpm at various test conditions.

6.0 VIBRATION CHARACTERISTICS

The vibration data was analyzed by computer analysis into the 1-, 2- and 4-per-rev components of the recorded waveforms. These frequencies are of primary interest for the analysis of two-bladed rotor vibration.

7.0 AIRSPEED CALIBRATION

The ship's standard airspeed system was calibrated by using the trailing bomb method. The data obtained compared favorably with data obtained from an airspeed calibration performed by the contractor on a different helicopter with the same airspeed system.

APPENDIX III

TEST INSTRUMENTATION

1.0 INTRODUCTION

Test instrumentation listed below was installed by the contractor and various calibrations were spot-checked by USAAVNTA personnel.

1.1 PILOT AND ENGINEER'S PANEL

a. Longitudinal Cyclic Stick Position (Meter)

b. Lateral Cyclic Stick Position (Meter)

c. Collective Stick Position (Meter)

d. Directional Pedal Position (Meter)

e. Sensitive Rotor Tachometer

f. Airspeed

g. Altitude

h. Engine Differential Torque Pressure

i. Fuel Flow

j. Fuel Totalizer

k. Compressor Inlet Temperature

l. g Meter

m. Ambient Air Temperature

1.2 PHOTO PANEL

a. Airspeed

b. Altitude

c. Ambient Air Temperature

d. Engine Output Shaft and Rotor Speed (Dual Tachometer)

e. Engine Gas Producer Speed

f. Engine Differential Torque Pressure

1.3 OSCILLOGRAPH

a. Collective Stick Position

b. Longitudinal Cyclic Stick Position

c. Lateral Cyclic Stick Position

d. Directional Pedal Position

e. Pilot Seat Vertical Vibration

f. Copilot Seat Lateral Vibration

g. Copilot Seat Vertical Vibration

h. Roll Angle

i. C.G. Vertical Vibration

j. Pitch Rate

k. Pitch Angle

l. Yaw Rate

m. Engine Differential Torque Pressure

n. C.G. Lateral Vibration

o. Litter Station Vertical Vibration (right side)

p. Litter Station Lateral Vibration (right side)

q. Voltage

r. Main Rotor Azimuth

s. Event Mark

t. Roll Rate

v. Throttle Twist-Grip Position

APPENDIX IV

CONFIGURATION OF UH-1B/540 ROTOR HELICOPTER RECEIVED BY USAAVNTA

1.0 MAIN ROTOR HUB AND BLADE ASSEMBLY

The "door-hinge" rotor design featuring the flex beam hub is the means by which the 540 Rotor System attains a stiff chordwise or in-plane structure with a soft flapping or beam structure. A broad, flat steel plate replaces the standard UH-1B round hub spindle. The high in-plane stiffness permits the use of a large amount of tip weight without an increase in the chord oscillatory loads. The tip weight, in connection with the hub flexure, reduces the beam oscillatory load. This results in a dynamically balanced design which minimizes oscillatory stress levels and rotor induced vibrations.

The main rotor blade chord has been increased to 27 inches. The rotor remains at 44-foot diameter and features 10-degree blade twist. The airfoil section is NACA 9-1/3 percent which is thinner than the 12 percent used on all other UH-1 helicopters.

Each blade has a 35-pound trim weight and a 20-pound weight installed in the leading edge "C" spar section as shown in Figure 1.

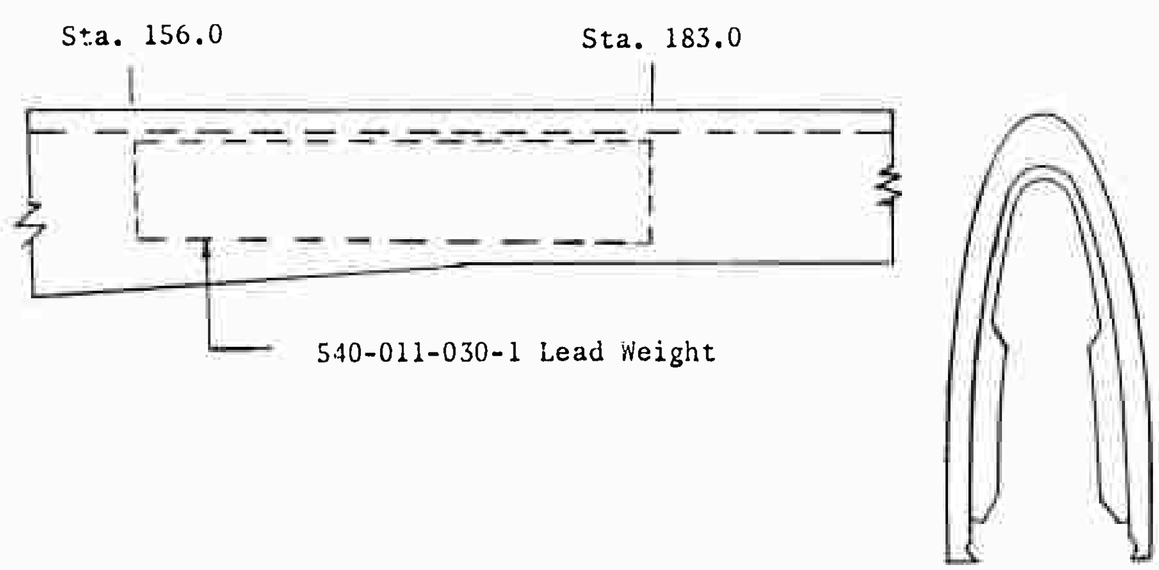


Figure 1

Figure 2 gives an exploded view of the new rotor head. Pitch changes to the rotor are achieved by motion inputs to the trailing edge pitch horns into the grips retaining the main rotor blades. The grip rotates on self-lubricated teflon bearings whose journals are positioned on each end of the yoke extensions.

Main rotor blade centrifugal force is transferred to the -102 yoke by means of the 204-012-112-7 wire straps housed within the -153 yoke extensions.

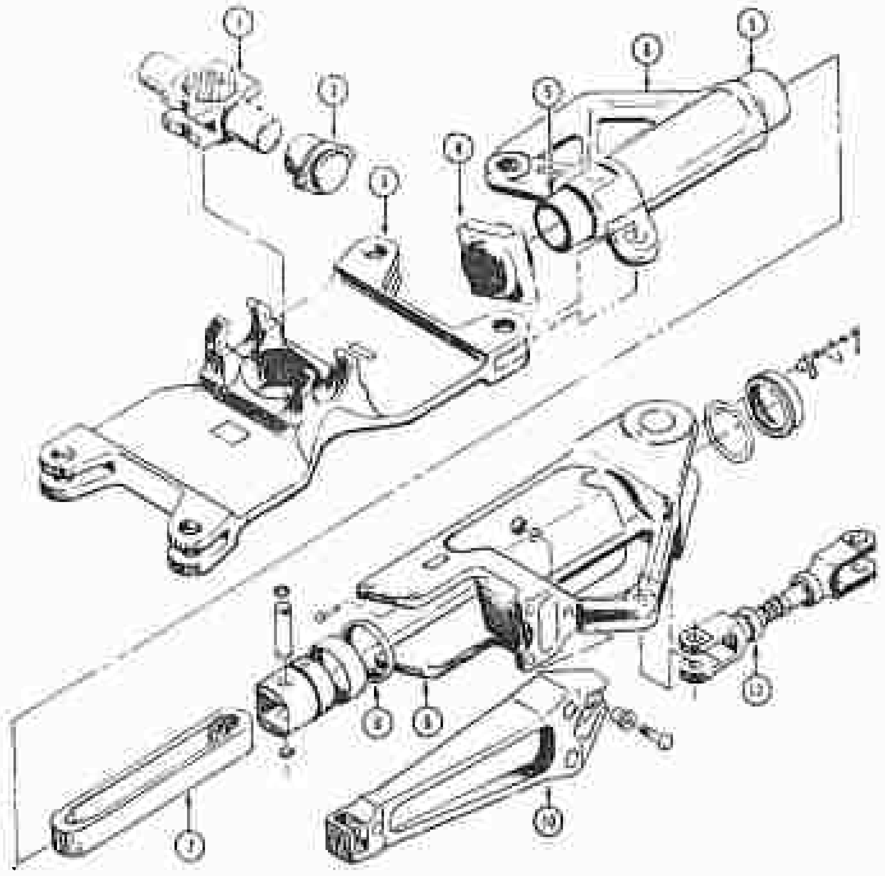


FIGURE 2. 540 Main Rotor Hub Assembly

1. 540-011-150-1 Trunnion
2. 540-011-106-1 Trunnion Housing Assembly (2reqd) (with dust seal)
3. 540-011-102-5 Yoke
4. Bearing Housing (with dust seal)
5. Pitch Change Bearing Journals
6. 540-011-153-1 Extension Assembly (2 reqd)
7. Tension Strap
8. Outboard Bearing Dust Seal (2 reqd)
9. 540-011-154-5 Grip Assembly (2 reqd)
10. 540-011-147-1 Pitch Horn Assembly (2 reqd)
11. 540-011-116-1 Drag Brace Assembly

Main rotor blade teetering motion is achieved by means of teflon bearings encased in the -106 housing and riding on the trunnion journals. These bearings, protected by a dust seal, need no lubricating fluid. Power is transmitted to the rotor by the splined trunnion.

2.0 ROTATING CONTROLS

The rotating controls are similar to standard UH-1B/D controls except that they have been appropriately strengthened to resist the higher control loads encountered at the increased airspeeds and gross weight limits established for the UH-1B with the 540 Rotor System.

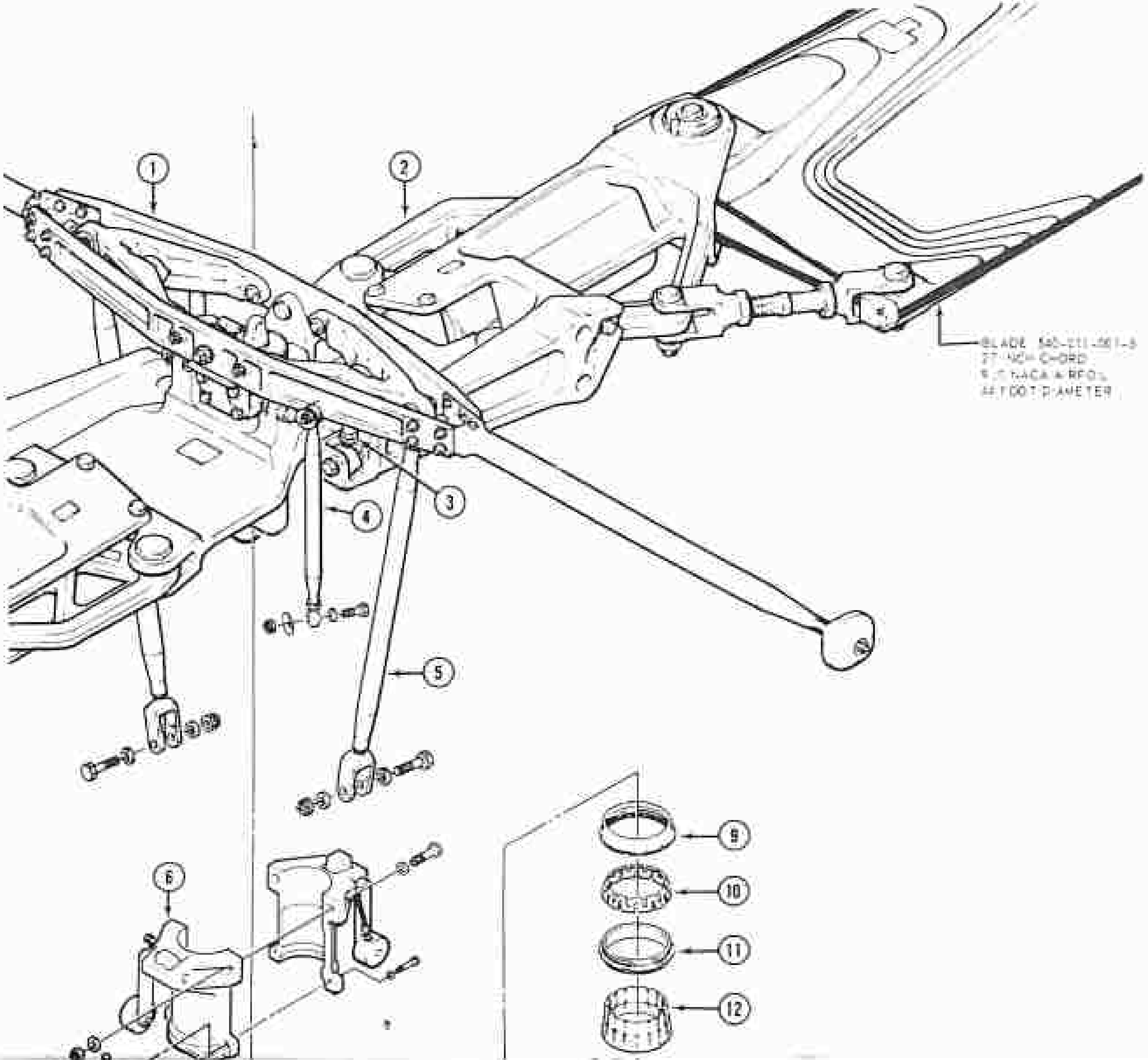
Figure 3 shows the control system detail components. The stabilizer bar is controlled by the same 204-010-937-5 dampers used on standard UH-1B/1D helicopters; however, the damper arm, 204-011-308-5, has been shortened.

A dynamic blade stop, -468, has been added to restrict minimum blade ground clearance at low rotor speeds (110 rpm and below) to assure safety of ground personnel.

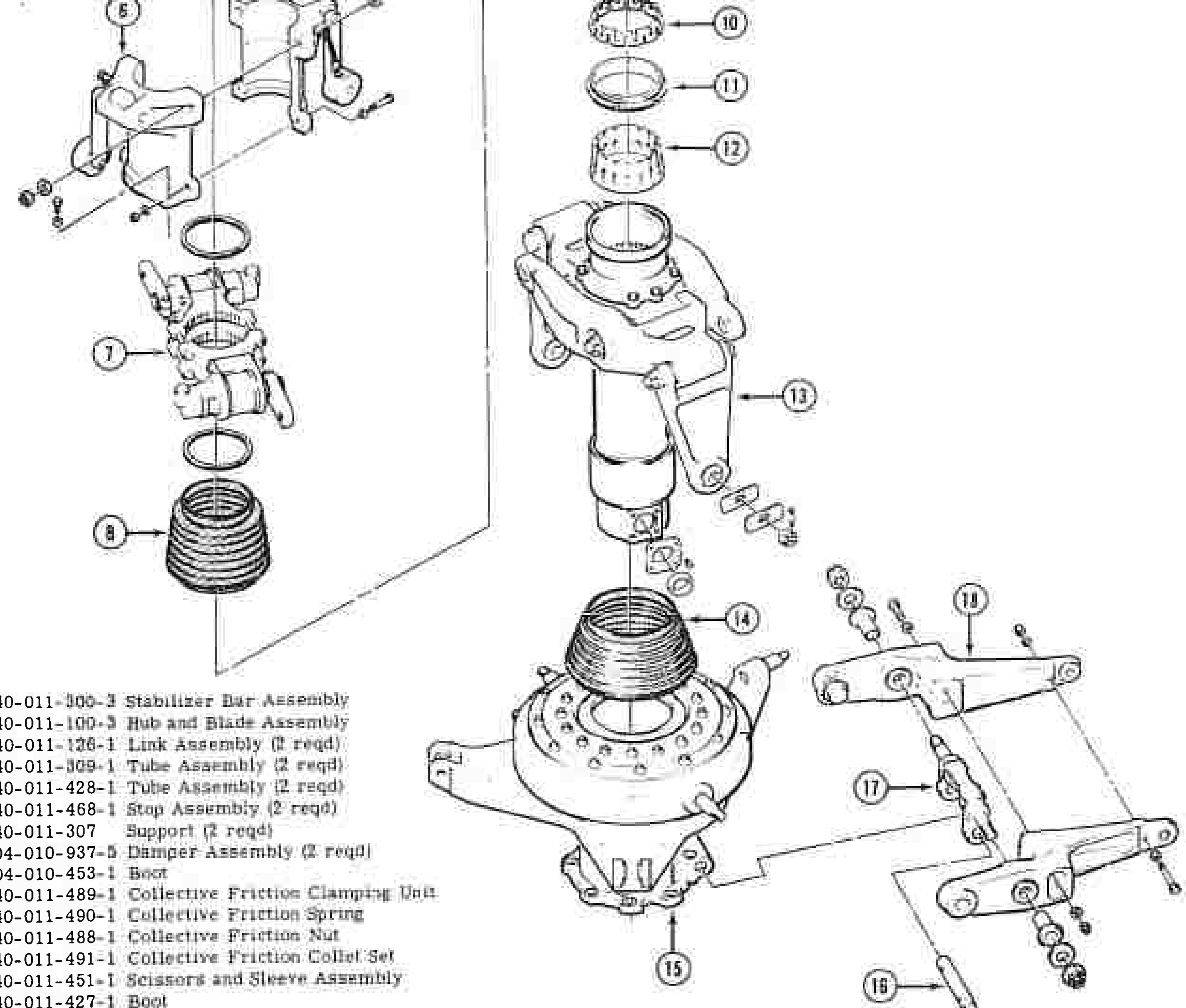
In addition, a collective friction device shown in Figure 3 has been installed to reduce helicopter 1/rev vibrations. This device includes the -491 collect set, which features a teflon bearing surface that is designed to ride on a stainless sleeve bonded to the transmission mast. The device is held in place with the -489 clamping unit on the -488 friction nut. Varying friction forces can be achieved by torquing the friction nut which bears against the -490 friction spring that transfers force to the bearing surface of the collect set. This spring is designed to compensate for bearing surface wear in such a way that a constant friction force should be maintained for the operating time between overhaul periods established for the rotating controls. Proper system friction is obtained by torquing the -488 nut until a 120-pound force is meaured at the control rod end of the -454 lever assembly.

Collective and cyclic motions are transferred to the main rotor through the -451 scissors and sleeve assembly. The -453 link assembly transfers -454 lever inputs into pure vertical motions of the collective sleeve assembly.

The -450 swashplate and support assembly features a swashplate which pivots through teflon bearings on a unibal in response to control inputs.



1
2
3
4
5
6
9
10
11
12
BLADE
27 INCH CHORD
NACA AIRFOIL
44 FOOT DIAMETER



1. 540-011-300-3 Stabilizer Bar Assembly
2. 540-011-100-3 Hub and Blade Assembly
3. 540-011-126-1 Link Assembly (2 reqd)
4. 540-011-309-1 Tube Assembly (2 reqd)
5. 540-011-428-1 Tube Assembly (2 reqd)
6. 540-011-468-1 Stop Assembly (2 reqd)
7. 540-011-307 Support (2 reqd)
   204-010-937-5 Damper Assembly (2 reqd)
8. 204-010-453-1 Boot
9. 540-011-489-1 Collective Friction Clamping Unit
10. 540-011-490-1 Collective Friction Spring
11. 540-011-488-1 Collective Friction Nut
12. 540-011-491-1 Collective Friction Collet Set
13. 540-011-451-1 Scissors and Sleeve Assembly
14. 540-011-427-1 Boot
15. 540-011-450-1 Swashplate and Support Assembly
16. 540-011-460-1 Pivot Pin
17. 540-011-453-1 Link Assembly
18. 540-011-454-1 Lever Assembly (2 reqd)

Figure 3. 540 Main Rotor Hub and Blades and Rotating Controls

2

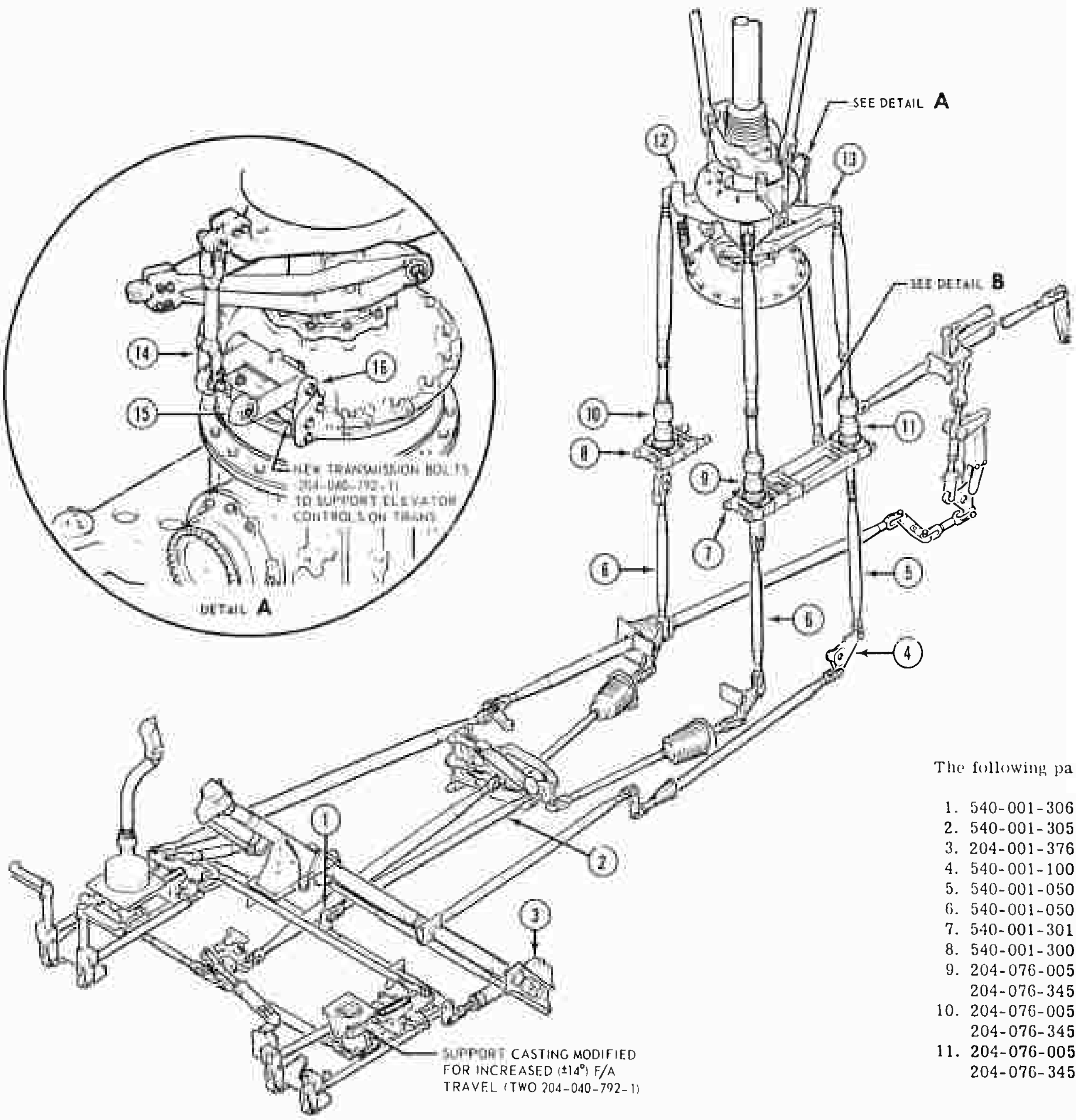
3.0 NON-ROTATING CONTROLS

The major changes in the non-rotating control system were incorporated to accommodate the increased cyclic travel required (±14 degrees) to obtain the higher air speeds as well as eliminate interference problems caused by changes in the collective sleeve/collective "A" frame in the rotating control system. Figure 4 shows the basic UH-1B control system with the required changes for the 540 Rotor System.

The cylinder support castings were replaced by increased strength forgings. P/N 540-001-300-3-301.

Control systems rigging instructions are given in 204-401-006.

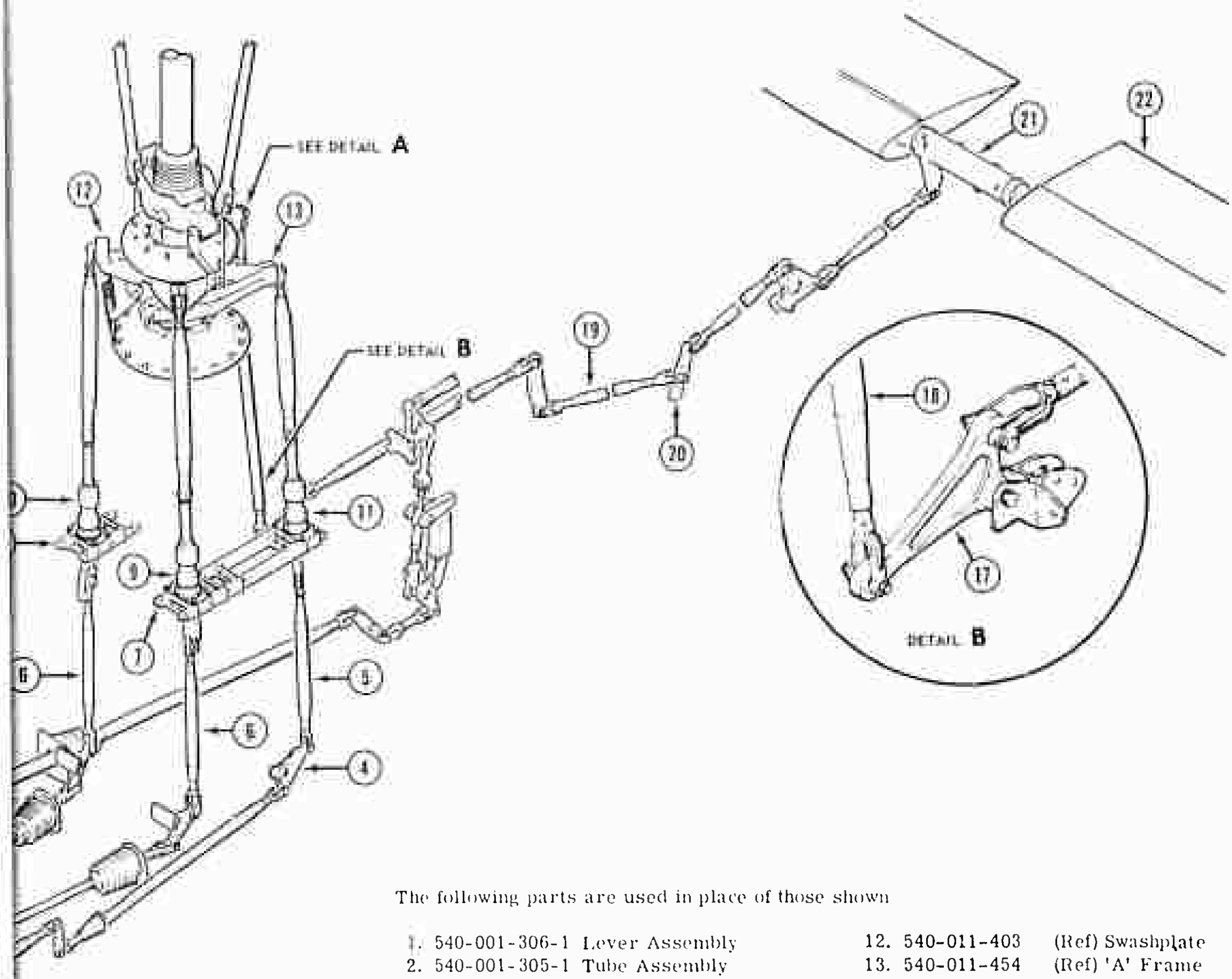
Not shown on Figure 4 are late changes made to improve cyclic stick trim. To achieve this, the 204-030-196 boots at Station 123 bulkhead on the 204-001-016-5 cyclic tubes were removed, a 2-1/2 pound counterweight was added to the 204-001-359-1 tube and lever assembly, and an increased rate spring was added to the 540-001-029-1 force gradient. When properly rigged, approximately 1.2 pounds/inch is required at the top of the cyclic stick with the force gradient system on boost system on in order to move the stick.



The following pa

1. 540-001-306
2. 540-001-305
3. 204-001-376
4. 540-001-100
5. 540-001-050
6. 540-001-050
7. 540-001-301
8. 540-001-300
9. 204-076-005
   204-076-345
10. 204-076-005
    204-076-345
11. 204-076-005
    204-076-345

Figure 4



The following parts are used in place of those shown

1. 540-001-306-1 Lever Assembly
2. 540-001-305-1 Tube Assembly
3. 204-001-376-3 Magnetic Brake
4. 540-001-100-1 Bellcrank Assembly
5. 540-001-050-1 Tube Assembly
6. 540-001-050-7 Tube Assembly (2 reqd)
7. 540-001-301-1 Support Assembly
8. 540-001-300-1 Support Assembly
9. 204-076-005-1 Cylinder Assembly
   204-076-345-1 Boot Assembly
10. 204-076-005-3 Cylinder Assembly
    204-076-345-1 Boot Assembly
11. 204-076-005-5 Cylinder Assembly
    204-076-345-1 Boot Assembly
12. 540-011-403 (Ref) Swashplate
13. 540-011-454 (Ref) 'A' Frame
14. 540-001-903-1 Link
15. 540-001-904-1 Lever Assembly
16. 540-001-905-1 Support Assembly
17. 540-001-907-1 Lever Assembly
18. 540-001-908-1 Tube Assembly
19. 540-001-910-5 Tube Assembly
20. 540-001-911-1 Lever Assembly
21. 205-001-914 Horn Assembly
22. 205-030-890 Elevator Installation

IFIED
/A
792-1

Figure 4.. Non-Rotating Control Changes for 540 System

2

4.0 TRANSMISSION ASSEMBLY

The transmission used for the 540 Rotor System is the same as the standard UH-1B transmission except that a 204-040-800 quill assembly is used to drive the dual dydraulic system pumps. The only other difference is that a pair of 204-040-792 bolts are used to attach the 540-001-905 elevator support casting to the transmission.

5.0 DUAL HYDRAULIC BOOST SYSTEM

This system features completely independent dual reservoirs, pumps, tandem servo-actuators, filters, switches, valves, pressure indicators and associated tubing and hydraulic lines. Both pumps are driven by a power takeoff from the transmission. Both systems power the main rotor, while System 2 actuates the antitorque boost cylinder for tail rotor control and System 1 actuates any armament system requiring hydraulic power.

The dual system operates under a 1500-psi pressure which results in a total load moving and reacting capacity of 2200 pounds at the servo-actuators. In the event of a failure in one subsystem, the second system maintains 1500-psi pressure and has a 1100-pound load moving capability at the servo-actuator. This, in conjunction with the 600-pound load reacting capability of the irreversible valves which are actuated only in the event one system fails, gives a total 1700-pound load resisting capability at the servo-actuator. The servo-actuator features two pistons on a single shaft.

From the module housings, the hydraulic fluid under pressure feeds into a manifold, then into ports in the servo-actuators. The 204-076-393 relief valve reduces system pressure to 1000 psi for the antitorque boost cylinder.

System 1 has a fluid capacity of 3.32 quarts while System 2 has a fluid capacity of 3.04 quarts. Dual system flow rate is 6.1 gallons/minute at 6600 engine RPM.

Prior to flight, the dual system can be checked by means of a three-position spring loaded toggle switch installed in a hydraulic panel in the upper portion of the console in the cockpit. Shown below are markings on this panel:

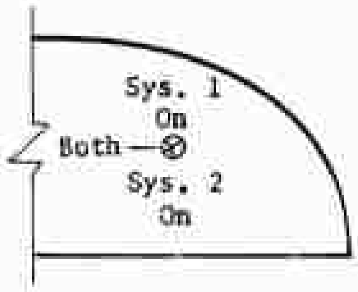


FIGURE 5

Both systems are normally on, and the test switch remains in a vertical position. Forward motion of the switch against a spring will deactivate System 2 with System 1 remaining on. When this action is taken, a capsule in the caution panel should indicate System 2 is out. Conversely, when the test switch is moved aft, System 1 is deactivated while System 2 remains on. The caution panel should indicate that System 1 is out.

6.0 TAIL BOOM

The tail boom for the 540 rotor system is the same as that for a standard UH-1D tail boom with exceptions. The air-foil section of the vertical fin formerly symmetrical has been changed to increase chord of the section and add camber. The leading edge of the fin remains the same. The trailing edge portion has been changed to accommodate honeycomb panels that achieve the camber effect. Figure 6 shows the new fin installation and a new lower vertical fin fairing. Also shown here is the UH-1D elevator installed on the tail boom.

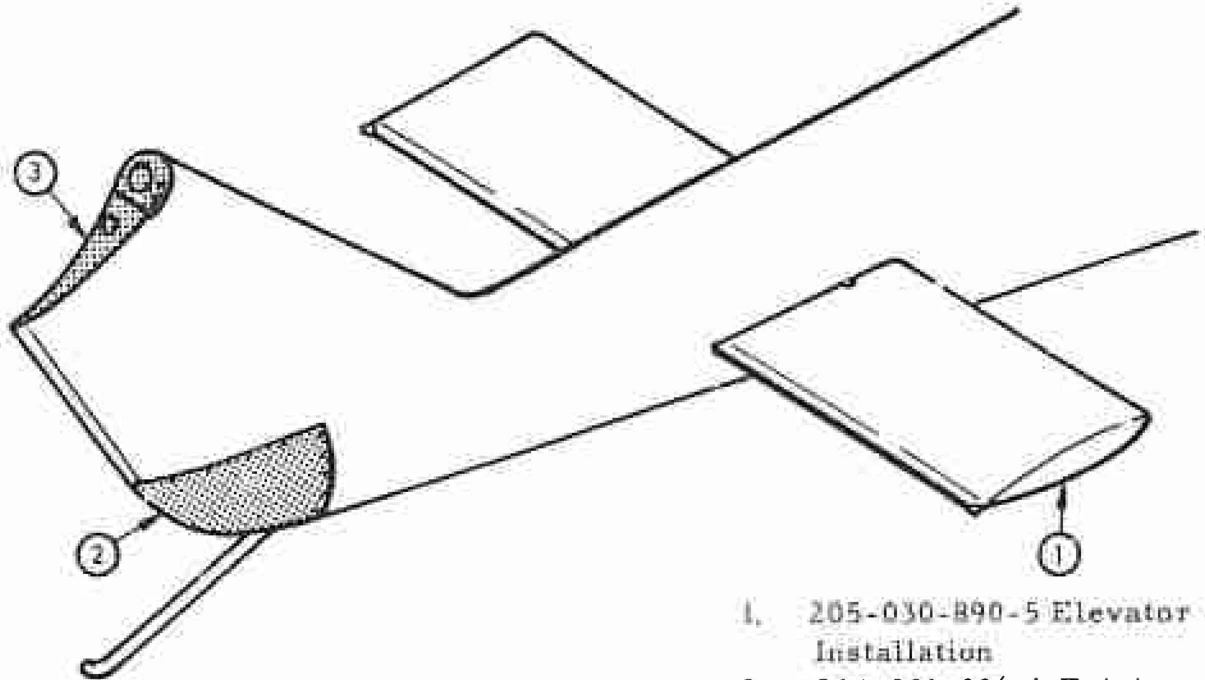


FIGURE 6. 540 Tail Boom

7.0 TAIL ROTOR HUB AND BLADE ASSEMBLY

In order to withstand the higher tail rotor assembly loads normally encountered at higher airspeeds, a modified hub assembly was incorporated into the production configuration of the 540 rotor system. This hub is similar to that used on standard UH-1B/D helicopter except that the inboard bearing was replaced with a 204-011-714 thrust unit to reduce system chord loads.

The tail rotor blades are the same as standard except the phenolic tip block has been replaced by an aluminum block to add 1/2 pound of tip weight in order to reduce blade beam loads.

Forty-eight foot rotor UH-1D cross head and pitch links have been added to obtain 22-1/2-degree tail rotor pitch travel.

8.0 AIRSPEED SYSTEM

The standard UH-1B airspeed system featuring independent static and dynamic ports was replaced with an integral static-dynamic pitot tube, 204-072-195, located on the cabin roof as shown in Photo 2.

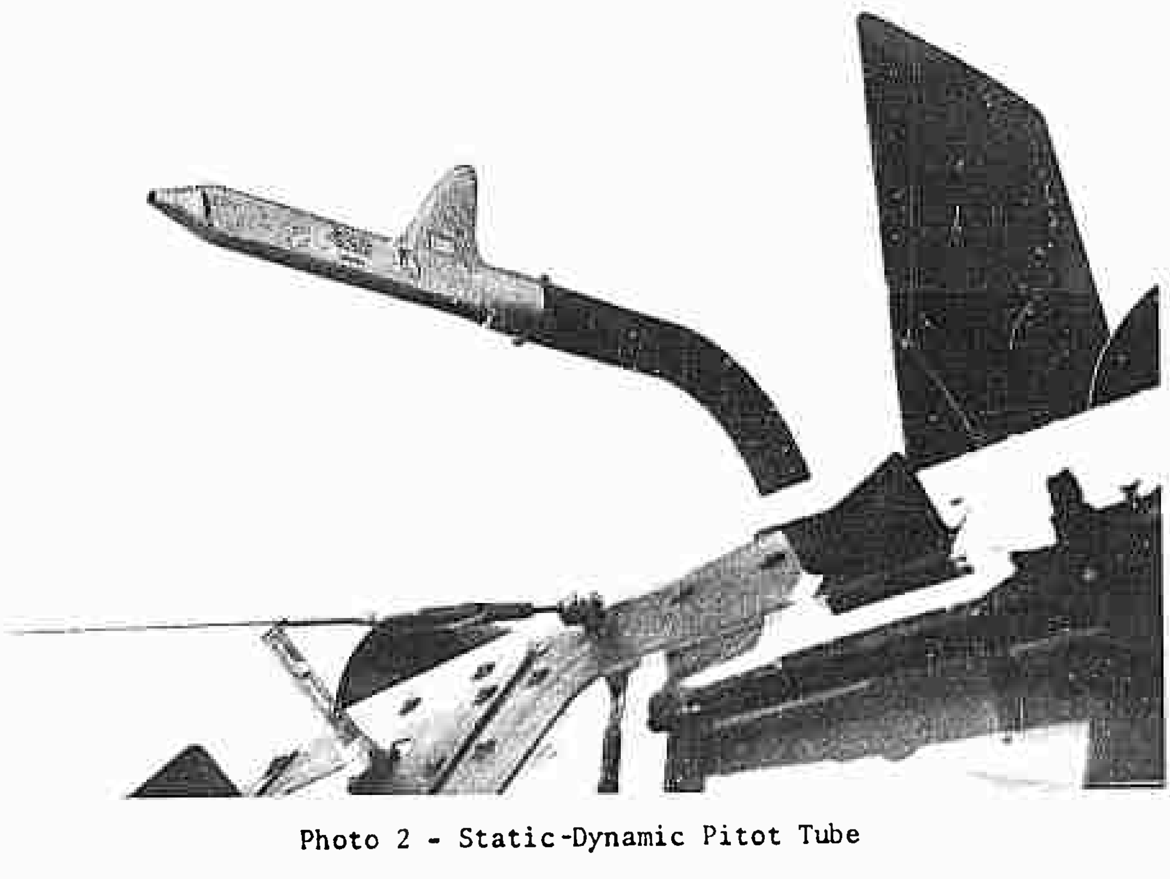


Photo 2 - Static-Dynamic Pitot Tube

APPENDIX V

UH-1B/540 ROTOR HELICOPTER PHASE C TEST RESULTS

1.0 PHASE C TEST

1.1 OBJECTIVE

Phase C tests were conducted to determine the suitability of the contractor's corrections to problems uncovered during the Phase B evaluation of the UH-1B/540 rotor helicopter. Primary emphasis was placed on the corrections for self-excited "pylon rock" and static longitudinal stability.

1.2 METHOD

Three flights were flown for a total of 5 hours productive flight time. The loadings for these flights were:

a. 6680 pounds, Aft C.G. (Station 137.5)

b. 8500 pounds, Mid C.G. (Station 131.0)

c. 9500 pounds, Fwd C.G. (Station 127.5)

Test instrumentation was not installed for Phase C test; therefore, Phase C results are based on pilot qualitative comments.

1.3 RESULTS

1.3.1 "Pylon Rock"

It appears that the self-excited "pylon rock" problem has been solved. USAAVNTA personnel were not able to obtain "pylon rock" as experienced in Phase B. The contractor determined the cause of self-excited pylon rock to be high control loads that produced deflection of the boost tube support structure. The deflection caused boost cylinder pilot valve action that resulted in erratic control inputs to the rotor. The contractor's corrective actions to solve self-excited pylon rock were:

a. Installation of production 540 rotor boost tube support structure that was stiffer than the non-production support structure used during Phase B.

b. Installation of boost tube pilot valve balance springs to prevent erratic action of the pilot valve.

1.3.2 Static Longitudinal Stability

Stick-free static longitudinal stability (force gradient with change in airspeed) at high speed was improved and found to be satisfactory. The stick-fixed longitudinal stability (position gradient with change in airspeed) at high speed was improved slightly and found to be satisfactory primarily because of the improved stick-free stability. The unsatisfactory stick-free static longitudinal stability experienced during Phase B was caused by unbalanced forces in the longitudinal control system that tended to overpower the cyclic trim spring system. The contractor corrective actions to improve static longitudinal stability were:

a. Installation of a stronger fore and aft cyclic force trim spring (1.5 pounds/inch in lieu of 1.0 pounds/inch to balance longitudinal control system forces.

b. Installation of a bob weight on the cyclic jack shaft located just aft of the cyclic stick to counterbalance the longitudinal cyclic control system.

c. Removal of fuel vapor rubber boots from the cyclic controls at Station 115.0 where the controls passed through the aft bulkhead. These boots acted as springs and contributed to the unbalanced forces of the longitudinal control system.

d. Reprogrammed elevator travel with forward cyclic control displacement to improve the position gradient at high speeds.

1.3.3 Trim Characteristics

The corrective action taken to improve static longitudinal stability also improved the trim characteristics at high speeds. The positive force gradient made it possible to depress the trim button on the cyclic stick without producing an associated stick movement that would have changed trim speed.

1.3.4 Autorotation

The rerigging of the elevator with forward cyclic control displacement has had little or no effect on the undesirable cyclic trim change required following a throttle chop.

Throttle chops at V_{max} with a flare to 60 knots were executed 400 feet above the ground. The chop and flare were accomplished without loss of altitude. This was submitted as a correction to Phase B comments.

1.3.5 Rotor Blade Tracking

Rotor blade tracking prior to Phase C required approximately three days and was still not optimum; consequently, the overall ride had deteriorated. The increase in 1-per-rev vibration was unsatisfactory and USAAVNTA personnel requested that it be improved. The contractor solved this problem by matching one installed blade with one from a spare set. Tracking of this set of blades was a much smaller problem and the time required was approximately four hours. The ride obtained with this set of blades was satisfactory and compared favorably with the ride obtained during Phase B testing.

1.3.6 Control Loads

The fixes incorporated to improve static longitudinal stability improved cyclic control harmony to the point where it was satisfactory.

During Phase B undesirably heavy collective control forces for hovering and cruising flight were present. The contractor is incorporating an 8-10 pound minimum collective friction level that is satisfactory for cruising flight but appears a little too heavy for extended hovering.

1.3.7 Pilot Induced Pylon Motion

The fixes incorporated to solve the self-excited "pylon rock" problem also improved slightly the pilot induced pylon rock. The vibratory response following a control pulse was slightly improved especially in the low-amplitude, lightly damped residual vibration experienced during Phase B testing.

1.4 Conclusions

The UH-1B/540 rotor helicopter, S/N 63-8684, is NOW ~~not~~ acceptable for Phase D testing.

1.5 Recommendations

The balance springs installed on the boost cylinder pilot valves to help cure the "pylon rock" problem have led the contractor to plan more investigating in this area to determine permissible valve overlap. It is recommended that the Project Manager, through the Army office at the contractor's facilities, monitor future results of this work.

The rotor tracking problem experienced during Phases B and C was attributed to the fact that these were the first 540 rotor blades built and not necessarily representative of future production hardware. It is recommended, therefore, that the Project Manager monitor rotor blade tracking problems encountered in future production UH-1B/540 rotor helicopters.

APPENDIX VI

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